See N-41416

Forty-second Annual Report

of the

National Advisory Committee for Aeronautics

1956

ADMINISTRATIVE REPORTS
WITHOUT TECHNICAL REPORTS

FORTY-SECOND ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

1956

ADMINISTRATIVE REPORT
WITHOUT TECHNICAL REPORTS

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Letter of Transmittal

To the Congress of the United States:

In compliance with the provisions of the act of March 3, 1915, as amended, establishing the National Advisory Committee for Aeronautics, I transmit herewith the Forty-second Annual Report of the Committee covering the fiscal year 1956.

DWIGHT D. EISENHOWER.

THE WHITE HOUSE,
JANUARY 28, 1957.

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Letter of Submittal

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON, D. C., *October 17, 1956.*

DEAR MR. PRESIDENT:

In compliance with the act of Congress approved March 3, 1915, as amended (U. S. C. title 50, sec. 151), I submit herewith the Forty-second Annual Report of the National Advisory Committee for Aeronautics for 1956.

Aeronautics is progressing at a remarkable rate. Scientific problems are multiplying in number, difficulty, and cost. Under military stimulation, development efforts are extending beyond our basic knowledge. Real progress can be accelerated, and on a more economical basis were scientific research accorded priority and conducted on an adequate basis in advance of development.

There is a growing demand for more scientists and engineers. Long range measures are being taken to stimulate the education of larger numbers. In the meantime, the law of supply and demand is forcing salaries up. Government research organizations, operating under rates of compensation fixed by law, cannot recruit or retain adequate numbers of scientists and engineers. NACA is losing outstanding and irreplaceable leaders in aeronautical science. This weakening trend must be reversed. The simplest and the best remedy is the enactment of legislation authorizing the Government to pay the going rates for scientists and engineers. Leadership in aeronautical science and American supremacy in the air are at stake. The necessary legislation is strongly recommended.

Respectfully submitted.

JEROME C. HUNSAKER,
Chairman.

THE PRESIDENT,
The White House, Washington, D. C.

National Advisory Committee for Aeronautics

Headquarters, 1512 H Street NW., Washington 25, D. C.

Created by Act of Congress approved March 3, 1915, for the supervision and direction of the scientific study of the problems of flight (U. S. Code, title 50, sec. 151). Its membership was increased from 12 to 15 by act approved March 2, 1929, and to 17 by act approved May 25, 1948. The members are appointed by the President and serve as such without compensation.

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FORTY-SECOND ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON, D. C., *October 17, 1956.*

To the Congress of the United States:

In accordance with Act of Congress, approved March 3, 1915, as amended (U. S. C. title 50, sec. 151), which established the National Advisory Committee for Aeronautics, the Committee submits its Forty-second Annual Report for the fiscal year 1956.

The airplane looms as the instrumentality that has changed previous concepts of military power and the course of history. In addition to the airplane, the missile has become a major factor in warfare. In the emergency created by the present international situation, the United States is expending unprecedented sums for the production of aircraft and missiles, on the effectiveness of which the security of the Nation may largely depend.

Numbers of aircraft and missiles alone are insufficient unless their performance is at least equal to those they may be called upon to oppose. This makes it essential to choose the most advanced designs for production, but allows little time to prove the new features incorporated. It falls to the aeronautical laboratories not only to provide the new ideas necessary to insure superior performance, but at the same time to prove in advance the soundness of the design as a whole. The Committee's work, therefore, falls into two principal categories; namely, research to furnish new ideas; and the application of those ideas to current military designs in cooperation with industry. The present emergency has naturally revised the priorities in connection with the long-range research program, to the end that those things which give most immediate promise are emphasized.

Only continued scientific research, on a scale adequate to meet growing needs, can give the Nation assurance that its aircraft and missiles will be kept at least the equal of those of any other nation. In order to develop them to their full potentialities, both in peace and in war, scientific research must be prosecuted with vigor and imagination.

In the last fifteen years, the speed of tactical aircraft has been increased from less than 400 to more than 1,000 mph. Through the use of special research airplanes, we have been able to penetrate the so-called sound barrier to the point where our research airplanes are flying faster than 2½ times the velocity of sound. We see the day, not too distant, when man can fly to any point on the globe in but a few hours.

To satisfy military requirements we must learn how to project missiles at thousands of miles per hour, along ballistic trajectories to targets far across the seas. At the same time, we are striving for the knowledge that will make possible satellites probing into regions beyond the earth's atmosphere and obtaining valuable information.

For the ballistic missile, a temperature of many thousands of degrees—higher than that on the surface of the sun—will be generated in the air near the surface of the missile. Under such conditions, the air molecules dissociate or split apart into their constituent atoms, and electrons are knocked out of atoms to make the air ionized and electrically conducting.

We need to duplicate in the laboratory the strange and difficult conditions of future flight, so that practical solutions of these problems can be found. Recently it has become possible to make small, pilot models with which to prove the practicability of constructing the expensive new tools of research necessary for the extension of the present limits of our knowledge.

But laboratory equipment for the experimental study of aerodynamic heating and other complex flight problems is only one requirement. More teams of talented young men competent to work in the new scientific fields are urgently needed. This requires a realistic approach to the pay problem. The attractiveness of public service has been critically depreciated by private industry offering salaries and "fringe benefits" far greater than the NACA is allowed by law to pay. We do not quarrel with the salaries paid by industry. We must, however, not only provide additional fringe benefits, but also offer rates of compensation sufficient to recruit, and to hold, the scientists and engineers necessary to perform the fundamental research that governs progress.

Over the years the career scientists of the NACA have made large contributions to the advancement of aeronautics, yet the cost has been very small when compared with the value of the results. Costs are increasing. Further advances in the art and science at a greatly accelerated rate are essential to our national security. We are confident that the National Advisory Committee for Aeronautics can continue to show the way to important advances in aeronautics if the Congress will provide the required support.

Respectfully submitted.

JEROME C. HUNSAKER,
Chairman.

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Part I—TECHNICAL ACTIVITIES

THE NACA—WHAT IT IS AND HOW IT OPERATES

During the 41 years since the Congress founded it as an independent Federal agency, the National Advisory Committee for Aeronautics has sought to assess the current stage of development of aircraft, both civil and military; to anticipate the research needs of aeronautics; to build the scientific staff and unique research facilities required for these research needs; and to acquire the needed new knowledge as rapidly as the national interest requires.

By discharging its primary responsibility—scientific laboratory research in aeronautics—the NACA serves the needs of all departments of the Government. The President appoints the 17 unpaid members of the Committee, who report directly to him. They establish policy and plan the research to be carried out by the 7,900 scientists, engineers, and other persons who make up the staff of the agency.

The NACA research programs have both the all-inclusive, long-range objective of acquiring new scientific knowledge essential to assure United States leadership in aeronautics and the immediate goal of solving, as quickly as possible, the most pressing problems. In this way, they effectively support the Nation's current aircraft and missile construction program.

Most of the problems to be studied are assigned to the NACA's research centers. The Langley Aeronautical Laboratory in Virginia works on structural, general aerodynamic, and hydrodynamic problems. The Ames Aeronautical Laboratory in California concentrates on high-speed aerodynamics. The Lewis Flight Propulsion Laboratory in Ohio is a center for power-plant studies. At the High-Speed Flight Station in California special fully instrumented research aircraft probe transonic and supersonic problems in flight. The Pilotless Aircraft Research Station at Wallops Island, Virginia, is a branch of the Langley Laboratory where rocket-powered free-flight models are used to attack aerodynamic problems in the transonic and supersonic speed ranges.

A major task of the NACA since its beginning in 1915 has been coordinating aeronautical research in the U. S. Through the members of the Committee and its 28 technical subcommittees, the NACA links the military and civil government agencies concerned with flight. The aviation industry, allied industries, and scientific institutions are also represented.

Assisting the Committee in determining and coordinating research programs are 4 major and 24 subordinate technical committees with a total membership of nearly 500. Members are chosen because of technical ability, experience, and recognized leadership in a special field. They also serve without pay, in a personal and professional capacity. They furnish valuable assistance in considering problems related to their technological fields, review research in progress at NACA laboratories and in other establishments, recommend new research to be undertaken, and assist in coordinating research programs.

Members of the technical committees and subcommittees and of the Industry Consulting Committee are listed in Part II of this report, beginning on page 80.

Research coordination is also accomplished through frequent discussions by NACA scientists with the staffs of research organizations of the aircraft industry, educational and scientific institutions, and other aeronautical agencies. Through a west coast office the NACA maintains close liaison with aeronautical research and engineering staffs in that important aviation area.

The NACA sponsors and finances a coordinated research program at 33 nonprofit scientific and educational institutions, including the National Bureau of Standards. In this way scientists and engineers whose skills and talents might otherwise not be available contribute importantly to Federal aeronautical research.

During the fiscal year 1956, the following institutions participated in NACA contract research:

National Bureau of Standards
University of Alabama
Battelle Memorial Institute
Polytechnic Institute of Brooklyn
Brown University
California Institute of Technology
University of California
Carnegie Institute of Technology
Case Institute of Technology
University of Cincinnati
Columbia University
Cornell University
Franklin Institute
Forest Products Laboratory
Georgia Institute of Technology
Johns Hopkins University
University of Kentucky
Lightning & Transients Research Institute
Massachusetts Institute of Technology
University of Michigan

University of Minnesota
New York University
University of North Carolina
University of Oklahoma
Purdue University
Syracuse University
University of Washington
University of Wisconsin
Southwest Research Institute
Stanford Research Institute
Stanford University
Stevens Institute of Technology
Yale University

Proposals from such institutions are carefully weighed to assure best use of the limited funds available to the NACA for sponsoring research outside its own facilities. Published research reports of the useful results of this part of the NACA program are distributed as widely as other NACA publications.

During the fiscal year, most of the NACA technical subcommittees reviewed research proposals from outside organizations or gave attention to research reports of completed contracts. There were 43 sponsored-research reports released during fiscal 1956.

Most of NACA's research information is distributed by means of its publications. Technical Notes and Reports are not classified for military security reasons and are available to the public in general. Translations of important foreign research reports appear as Technical Memorandums. The NACA also prepares research reports containing classified information. For reasons of national security, these receive carefully controlled circulation. When such information can be declassified, the research reports may be given wider distribution. Current NACA publications are announced in the NACA Research Abstracts.

Every year the NACA holds a number of technical conferences with representatives of the aviation industry, the universities, and the military services present. Attendance at these conferences is restricted because classified material is presented and the subject matter discussed at each conference is focused on a specific field of interest.

BOUNDARY-LAYER CONTROL

"It is established by reliable experiments that fluids like water and air never slide on the surface of the body; what happens is the final fluid layer immediately in contact with the body is attached to it (is at rest relative to it), and all the friction of fluids with solid bodies is therefore an internal friction of the fluid. Theory and experiment agree in indicating that the transition from the velocity of the body to that of the stream in such a case takes place in a thin layer of the fluid, which is so much the thinner, the less the viscosity. In this layer, which we call the boundary layer, the forces due to viscosity are of the same order of magnitude as the forces due to inertia. . . ."

Publication 35 years ago of the NACA document containing the above definition of the boundary layer (Report 116, "Applications of Modern Hydrodynamics to Aeronautics," by Ludwig Prandtl, 1921) together with issuance of Glauert's volume, "Aerofoil and Airscrew Theory," in 1926 has been credited by Sir William S. Farren with having "revolutionized our understanding and powers of analyzing the main phenomena which determine the lift and drag of aircraft, the thrust and torque of propellers, and the interference of wind tunnel constraints. It is hardly too much to say that a mass of indigestible 'facts' was rendered of historical interest by a few pages of inspired theoretical work." Even earlier (in 1904 in a paper read to the Third Congress of Mathematicians at Heidelberg) Prandtl had laid down the bases of the boundary-layer theory.

For more than a half century, then, the very complicated mechanics of the boundary layer have been studied. Especially at high lift, high pressure gradients exist on the upper surface of a wing, that is, regions where the upper surface pressures increase rapidly in a positive sense toward the wing trailing edge. Such a gradient will exist just aft of the wing leading edge and at the forward edge of a deflected flap.

As boundary-layer air traverses these pressure gradients, the pattern of velocity distribution within the boundary layer changes from one in which velocity increases rapidly from zero at the surface to a free-stream value just above the surface to a pattern in which for a short distance above the surface velocity is zero, and thence to a pattern in which flow reverses in a region just above the surface and goes toward the wing leading edge. This condition arises when the air next to the surface has not enough dynamic energy to move in the direction of the wing trailing edge against the increasing pressure of the stagnant layer.

With the occurrence of reverse flow, the whole flow mass breaks away, or separates, from the upper surface and lift is sharply reduced. The pressure gradient at the leading edge increases with increase in angle of attack; when the pressure gradient becomes so steep that the boundary layer separates from the airfoil surface, further increase in angle of attack produces no increase in lift, and frequently produces a significant loss.

Aerodynamicists recognized early that gains of increased lift and reduced drag could be made by preventing or delaying this separation of the air flowing, over the upper surface of the wing. Suggested ways to provide a measure of boundary-layer control included slots and flaps actuated by the pressure difference between the upper and lower surface of the wing, served as "natural" pumps by providing a flow of air that delayed separation.

What is believed to have been the first use of such a device came soon after World War I. Gustav Lachmann in Germany and Sir Frederick Handley-Page in England, working independently at first and then together, developed leading edge slots. These opened when the wing was inclined to a high angle of attack, thus preventing flow separation near the leading edge. Another device, developed for delaying separation and installed on airplanes in later years, is the slotted flap near the trailing edge of the wing.

In the late thirties, the NACA developed airfoils which provided laminar (non separated) flow to a degree far greater than that previously obtainable.

This work of changing the shape of the airfoil, performed at the Langley Aeronautical Laboratory, made it possible to delay the transition from smooth, laminar flow to turbulent or eddying flow to such an extent that the skin friction, or basic air resistance, of a wing could be reduced by almost two-thirds. When first adapted for use, on the North American P–51 Mustang, the so-called laminar-wing principle enabled design of a high-speed fighter with superior long-range capabilities.

Another way of postponing the transition to turbulent air flow through boundary-layer control is that of using mechanical pumps to suck in air and thereby limit the growth of the boundary layer. By 1940, study of this aspect of the boundary-layer-control problem had begun at the Langley Laboratory, and by the early years of World War II the work had progressed to full-scale flight testing.

In 1927, to cite an example of early research on the use of mechanical methods to provide boundary-layer control for high lift, a Navy trainer, a Curtiss TS–1, was modified at the Langley Laboratory for flight study of the system of boundary-layer control by blowing which Richard Katzmayr of Vienna had suggested. Two spanwise slots were installed on the upper surface of the upper wing to blow away the boundary-layer air. Blowing air was supplied by a conventional supercharger.

In the following 20 years, wind-tunnel investigations repeatedly demonstrated that large aerodynamic gains could be obtained from use of boundary-layer control. NACA Report 385, published in 1931, presented results showing that the maximum lift coefficient could be increased as much as 96 percent. Although large aerodynamic benefits were thus demonstrated, there was little enthusiasm for adapting such boundary-layer control schemes to airplanes of those days. Not only was the weight of the required pumping equipment considered prohibitively high, but, for airplanes of low wing loading, the use of large span flaps usually provided lift coefficients sufficiently high to permit acceptably low landing speeds.

In the middle forties, the design changes required to enable airplanes to reach supersonic speeds (as well as very low flight speeds) made development of useful, mechanical boundary-layer-control systems which could produce high lift a matter of greatly increased interest. Too, the tremendous increase in power available in the newly developed turbojet engines offered attractive possibilities for the needed supplies of suction or blowing air.

Among factors limiting the high-lift capabilities of modern high-speed airplanes have been the following: (1) Reduced wing aspect ratio and increased wing sweep, which reduce wing lift at a given angle of attack and also reduce the lift increment produced by flap deflection; (2) wing-thickness reduction which lowers the stalling angle of attack, and thus limits maximum lift, and (3) reduction in wing stiffness, requiring that ailerons be moved inboard to avoid aileron reversal at high speeds, leaving little or no wing area to convert to flaps.

Boundary-layer control is applied to trailing-edge flaps to achieve all the lift that theory indicates is available. If the increase in lift with increasing flap deflection for any given wing with conventional flaps is compared with that under the ideal conditions of theory, it is found that even for flap deflections as low as 20° the lift increment falls short of the theoretical value. At flap deflections as high as 60°, the lift increment may be only half that suggested by theory. If the flaps could be made to act as theory predicts, a substantial increase in lift could be realized on any flapped wing, without changes in other design features.

Proper application of boundary-layer control to the flaps will result in approaching the theoretical values of lift up to flap deflections of 60° to 70°. This has been demonstrated on a wide variety of wings ranging from those typical of fighters (thin, low aspect ratio, and high sweep) or bombers and jet transports (moderate thickness, high aspect ratio, and moderate sweep) to those of slow-speed airplanes (thick, high aspect ratio, and no sweep).

Design details differ, of course, but the prime objective was realized in each of these cases. An interesting corollary of these results is that with the flap effectiveness maintained to high flap deflections, the function of flaps and ailerons can be combined in a single unit without a major sacrifice in the effectiveness of either.

Research has demonstrated that boundary-layer control applied to a wing leading edge or to a wing leading-edge flap can greatly extend the useful angle-of-attack range. Boundary-layer control of the wing leading edge has also been successfully applied to a variety of airplane types.

Again, design details must differ because of differences in airplane design. For example, the application of boundary-layer control to the wing leading edge can have powerful and differing effects on the longitudinal stability of the airplane, depending on the location of the horizontal tail.

It is believed, however, that sufficient research has been completed on the application of boundary-layer control to a wing leading edge to enable the designer to use it with no more detailed "tailoring" for his specific design than would be required for a leading edge without boundary-layer control.

One point worthy of comment has been noted often in the course of boundary-layer-control studies. This is the need for more effort to be devoted to the problem of stall occurring at the wing leading edge as a result of trailing-edge flap effectiveness being increased. It was found that with boundary-layer control applied to trailing edge flaps of wings incorporating conventional slats at the leading edge the greater lift due to boundary-layer control results in the slats being unable to delay leading-edge stall to the desired angles of attack. In other words, although the flap lift increment of the basic design was increased a significant amount, the maximum total lift was limited by leading-edge stall. It may well be that the stall may be delayed sufficiently at the wing leading edge only through mechanical boundary-layer control.

It is important to note, then, that the apparently simple expedient of applying boundary-layer control to the trailing-edge flaps of an existing airplane may result in disappointing lift characteristics. Like many other facets of airplane design, satisfactory application of boundary-layer control cannot be realized by focusing attention on one phase only. The effects of boundary-layer control must be considered in the whole design.

In achieving boundary-layer control by removing the low-energy air, a theory was developed which predicted that if suction over one long slot, rather than through a number of separate slots could be realized, the total volume of air required would be reduced very greatly. Experiments showed that this result could be largely realized where use of sintered metals permitted approximately continuous suction. Although the flow volume needed to achieve boundary-layer control was not reduced to the level predicted by theory, it was brought down to a practical value when considered in terms of duct size and pumping equipment.

The range of experiments, using suction through sintered metal, covered applications to wing leading edges and leading-edge flaps and to wing leading-edge flaps on 10 different wings typical of turbojet fighters and bombers and propeller-driven transports. For example, on a North American F–86 airplane, area-suction boundary-layer control increased flap lift 50 percent at a flap deflection of 55°.

In this instance the required boundary-layer control was obtained with a 12-horsepower motor drawing 20 cubic feet of air per second through a 20-square-inch duct. Similarly, practical installations appear possible in other instances of area suction that have been examined.

Major research effort on blowing boundary-layer control has been directed toward reducing the mass flows required at high pressure ratios to a point where the use of flowing boundary-layer control would not compromise the engine or airplane operation. For example, the amount of bleed air required at the pressure ratios available during take-off must be low enough that the accompanying engine thrust loss does not cancel potential improvement in take-off characteristics resulting from the higher lift due to boundary-layer control. Also, the bleed air required at the pressure ratios available during approach and landing must be low enough that the engine is not operated at high valves of rpm and thrust simply to provide an air supply for the boundary-layer control.

The effectiveness of blowing-air boundary-layer control depends upon the thrust of the issuing jet. In other words, the same effectiveness can be obtained by discharging small quantities of air at very high velocities as with large quantities of air at relatively low velocities. This fact makes easier the matching of boundary-layer-control requirements with jet-engine characteristics.

The experiments and analyses made to date show that in every case studied, careful design based on the research data will enable these conditions to be met while realizing full benefits of the boundary-layer control. For instance, on a swept-wing airplane design typical of proposed light transports for bombers and powered by two Pratt & Whitney J–57 turbojet engines, full boundary-layer control on the trailing-edge flaps could be realized by use of 6 pounds of air per second at a pressure ratio of 5. This air quantity can be provided by one J–57 engine at idle conditions, without affecting its operation significantly. At higher engine thrust conditions, of course, the engine can supply considerably more pressure and air than is required.

Flight research on boundary-layer control has been carried on at the Ames and Langley Laboratories. Principal goals of this full-scale research, now concentrated at Ames, are three-fold: (1) to verify results which were demonstrated in the wind tunnels, (2) to gain firsthand experience with the practical problems of using boundary-layer control, and (3) to obtain both qualitative and quantitative measurements of pilots' reactions to the use of boundary-layer control.

At the Ames Laboratory, North American F–86 and FJ–3 airplanes have been modified and flown with area suction boundary-layer control applied to the wing leading edge and to the trailing-edge flaps and with blowing boundary-layer control applied to the trailing-edge flaps. In no case was it practicable to modify the structure to permit installation of a complete (both leading-edge and trailing-edge) system. In all cases, because the boundary-layer-control installation was not part of the basic, original design, the improvement in the overall performance characteristics fell short of what otherwise would have resulted.

Nevertheless, very useful results have been obtained and the program is being extended at Ames to include application of blowing boundary-layer control to both leading-edge and trailing-edge flaps on a North American F–100. In addition, NACA pilots have flown, and to varying degrees evaluated, most of the few airplanes modified by the aircraft industry to include boundary-layer control.

The study of pilot reaction to the use of boundary-layer control for high lift is being expanded into a more general study of the factors involved in choice of landing and take-off techniques. It has been noted that every pilot has taken advantage of some or all of the gain in lift brought about by the use of boundary-layer control, resulting in reduced landing speeds. Without exception, pilots have commented on the increased smoothness of flight and increased controllability resulting from boundary-layer control. The chosen approach speed of a North American F–86 in a GCA approach was dropped from 140 to 115 knots, and the pilot commented that even at the lower speed the airplane was easier to hold to the desired flight path.

Perhaps the most interesting, though as yet tentative, conclusion drawn from the flight-test results is that boundary-layer control is tending to remove lift as a limiting factor in choosing a minimum flight speed. In its place the pilots are stating that "altitude control" or "ability to check sink" is becoming the limiting speed factor. Exploratory studies indicate that the pilots are using engine thrust for this control and that their minimum speed choice is significantly affected by the minimum rpm at which adequate engine response can be obtained.

Lack of space in this report prevents consideration of a number of other interesting aspects of boundary-layer-control research by NACA scientists. Among them are the investigation of the whole area of boundary-layer control from inlet to exhaust of the power-plant system and the renewed efforts being made to learn how to employ boundary-layer control for drag reduction at very high speeds.

Today, the necessary basic research on the problems of boundary-layer control for high-lift conditions have been essentially completed. Now, it is possible to estimate with a fair degree of accuracy the gains to be realized in lift, the necessary cost in power required, and the design details associated with the use of boundary-layer control for a specific airplane. Information now in hand strengthens the belief that further effort to utilize boundary-layer control in future designs will be repaid by substantial performance improvement.

As wing loadings are increased to values beyond 100 pounds per square foot, entirely new concepts may be required to provide desired landing and take-off performance. Under study at the Langley Laboratory is the jet-flap principle, where a large percentage, or even all, of the primary power-plant jet exhaust thrust is used to augment lift. Exploitation of this idea may make it possible for the high-performance airplanes of the future, both civil and military, to take off and land at airports of present day proportions. Substantial application of the concept in the design of vertical- and short-takeoff-and-landing aircraft also appears likely.

HIGH-SPEED FLIGHT RESEARCH

From its conception in 1944 as a device for obtaining essential information about the problems of transonic and supersonic flight (information at the time mostly unobtainable in wind tunnels) the specially-designed, specially-instrumented airplane has become one of the most valuable tools of aeronautical research. So rapid and so spectacular have been the performance gains that it may be difficult to recall that, only 12 years ago, even the most advanced airplanes were limited by "compressibility" to Mach numbers below 0.85.

The first and most obvious benefit from the trail-blazing speeds and altitudes made by research airplanes was the confidence engendered by the demonstration in 1947 that flight through the fully unknown transonic range was possible. Almost overnight, supersonic flight by tactical aircraft became possible by straightforward means.

More important to the rapid performance improvement of our tactical airplanes has been the mass of detailed information resulting from intensive use of the research airplanes. To date, nearly 400 technical reports have been prepared by the NACA, covering flight-test results together with related wind tunnel investigations and theoretical studies. So rapidly has the information been assimilated into design practice that today one of the most challenging aspects of the continuing research-airplane program is the task of acquiring additional data fast enough to stay ahead of the new, high-performance airplanes now being designed and built.

Conduct of the research-airplane program has been a wholly cooperative venture, with the resources of the Air Force and the Navy, the participating airframe and engine manufacturers, and the NACA being used as needed. The military services have provided funds and generally have performed the mission of exploring and exploiting the maximum performance capabilities of the airplanes. The manufacturers, of course, have designed and built the airplanes and engines and in most instances, their pilots have carried out demonstration flights to satisfy contractual requirements. The NACA has furnished the latest research information for application in design of the airplanes, has made necessary wind-tunnel investigations using models of the airplanes, has provided data-gathering instrumentation for the planes, and has conducted the research flight programs.

What follows is a summary of this fruitful program, with special emphasis on the part the NACA has played.

Although the bulk of the NACA's research effort in the years of World War II was concentrated upon obtaining the quick-fix answers needed for performance improvement of airplanes already in production, it was obvious as early as 1943 that the very large amounts of thrust inherent in the turbojet and rocket engines coming into use offered the means of attaining supersonic flight. It was realized, however, that diligent and vigorous research effort using new tools and new techniques was essential—new aerodynamic knowledge had to be acquired before the supersonic speed potential could be exploited.

Existing knowledge of transonic air flow was pitifully small, and earlier efforts to develop a body of useful transonic theory generally had been unsuccessful. It was necessary to use experiment, but the principal tool of aerodynamic research, the wind tunnel, was subject to "choking" near the speed of sound and new techniques had to be devised. As victory loomed in Europe and the Far East, the NACA's effort in this direction was increased and broadened to include all approaches which offered promise.

Among the techniques employed were: (1) Specially instrumented aerodynamic bodies dropped from airplanes at high altitude; (2) small models placed on the wings of airplanes to capitalize upon the accelerated air flow above a curved surface, and (3) rocket-propelled models fired from the ground. With later advances in radar and radio-telemetering equipment, the rocket-propelled model has proved most valuable in research.

One of the earliest ideas for exploring the transonic speed range was suggested in 1943. Specially designed airplanes could be used to probe the transonic speed range. They could be propelled by the most powerful engines available; they could be loaded with data-recording instruments. Danger could be minimized if the research were performed at great altitude and in level flight; the loads imposed upon the structure would thus be kept low, and, if trouble developed, the pilot could throttle back to lower speed. For this conception of the research airplane, John Stack of the Langley Aeronautical Laboratory shared the 1947 Collier Trophy with the late Lawrence D. Bell, president of Bell Aircraft Corp. which built the X–1, and with then Capt. Charles E. Yeager, USAF, the first man to fly supersonically.

In 1944 contracts were let for construction of the first two research airplanes. Bell Aircraft Corp. began design of the X–1 (originally known as Project MX653 and later as XS–1) under Air Force sponsorship, and the Douglas Aircraft Co. about the same time undertook construction of the D–558, sponsored by the Navy. The first of these was to be powered by a rocket motor; the second, by a turbojet engine.

First to reach the flight-test stage was the X–1, of which two originally were constructed. The fuselage lines were adapted from the basic shape of a 0.50 calibre bullet. Straight wings, with thick, tapered aluminum skin, were provided to insure enough structural strength to withstand the loads expected at the altitudes and speeds programmed. The strength factor was specified at 18g, instead of the 12g then required for fighters. One set of wings had a thickness of 10 percent, much less than anything then flying; the second set was only 8 percent thick.

The power plant of the X–1 was a four-barrel rocket engine developed by Reaction Motors, Inc., under Navy contract. The engine used a simple but effective system of regenerative cooling (one of the liquid propellants was passed over the outer surface of the combustion chamber before being injected into the combustor) conceived in the midthirties by James H. Wyld, one of the founders in 1941 of Reaction Motors, Inc. Each rocket barrel produced 1,500 pounds of thrust, for a total engine thrust of 6,000 pounds.

Originally the R. M. I. rocket engine was to have incorporated a turbine pump to force the liquid-oxygen and alcohol-water fuel into the combustion chamber at a high rate. When development of a pump capable of operating at the required low temperatures impeded construction of the rocket engine, it was decided to use nitrogen under pressure to handle the fuel flow despite the resulting reduction in fuel carried.

In the fall of 1945, before the rocket engine was ready, the airframe of the X–1 was completed. To save time, Bell engineers proposed carrying the airplane aloft in a "mother ship," then releasing it to fly without power. In this way the general airworthiness of the X–1 could be determined before the rocket motor was completed. These tests were conducted early in 1946 at Pinecastle Air Force Base, Florida, and a group from the Langley Laboratory was sent to maintain and operate the 500 pounds of instrumentation carried by the X–1 and to provide technical guidance. The "mother ship" technique has been used since with others of the research-airplane series.

Following the successful glide tests it was decided that, in the interest of maximum safety for the pilot, powered flights should be made in the vicinity of the largest available landing area. The choice was the Air Force installation on the edge of Rogers Dry Lake in the Mojave Desert of California, known then as Muroc and now as Edwards Air Force Base.

After installation of the R. M. I. engine at Bell Aircraft's Niagara Falls plant, the first of two X–1 research airplanes was taken to Edwards early in October 1946. The previous month, 13 engineers, instrument technicians, and technical observers, all from Langley, were designated the NACA Muroc Flight Test Unit. On September 30 they began work at the desert base. The first successful rocket-powered flight was made December 9 by Chalmers H. ("Slick") Goodlin, company test pilot. By June 1947 performance up to a Mach number of 0.8 was fully demonstrated by Bell Aircraft pilots in a series of 21 powered flights.

On June 30, at a meeting at Wright-Patterson Air Force Base the Air Force and NACA agreed to divide responsibilities. Each agency was to use one of the X–1 airplanes in complementary flight programs. The Air Force objective was to exploit the airplane's maximum performance in as few flights as were reasonable, consistent with safety. The NACA program was necessarily more extensive: to acquire the desired detailed information. The NACA group, now permanently assigned at Edwards, was to furnish engineering and instrumentation assistance to the Air Force group, while the air launching of the NACA airplane was to be handled by the Air Force.

The Air Force received its X–1 in August 1947; mechanical difficulties delayed flights of the other model by the NACA until after the first of the year. Details of the historic first supersonic flight on October 14, 1947, by then Capt. Charles E. Yeager, USAF, are so well known as to need no recounting here. On March 4, 1948, NACA pilot Herbert Hoover became the first civilian to fly faster than sound.

Before turning attention to the research airplanes which came later, the following quotation from Maj. Gen. Albert Boyd, USAF, himself one of the first to fly supersonically, is given as a summation of the effectiveness of the cooperative program:

"The combination of talents served two purposes. First, the accelerated USAF program permitted a rapid exploration of the capabilities of the X–1 to the highest speed attained; and, secondly, the detailed NACA program provided the comprehensive data needed to develop complete envelopes of aircraft performance which might reveal unsatisfactory flight characteristics at some intermediate point. When considered separately, each program was a notable contribution to flight research, however, when combined, the results stand as a monumental tribute to both the USAF and NACA since the sonic barrier monster was not only completely licked, but a blow-by-blow account of its defeat was recorded for future use.

"The end results of high-speed flight-research programs conducted jointly made available to aircraft designers, for the first time in the history of flight testing, sorely needed information which served a dual purpose. The rapid but sketchy USAF portion of the program supplied answers which went toward determining the military applicability of a research aircraft, whereas the lengthy but detailed NACA program confirmed or refuted wind-tunnel data and at the same time provided information which would permit aircraft designers to avoid dangerous flight characteristics in future military and civilian aircraft of a more advanced design."

The Douglas D–558 research airplane project, sponsored by the Navy, quickly became a two-fold effort. The phase-one airplane, christened Skystreak, was similar to the X–1 in general layout. It was powered by a General Electric TG–180 turbojet instead of a rocket engine, and, because its potential was limited to transonic speed, the strength requirement was dropped from 18g to 12g.

Flown for the first time in mid-March, 1947, the Skystreak was used in manufacturer's performance demonstrations for about 5 months. On August 20 of that year the Navy Skystreak project officer, then Comdr. Turner Caldwell, flew the airplane to a new world's speed record, 640.7 mph. Five days later, then Maj. Marion Carl, USMC, added another 10 mph.

Three Skystreaks were built. One was destroyed on take-off in May 1948. The two other D–558–I airplanes were used intensively by the NACA in study of flight problems at speeds up to a Mach number of 0.9. As the program developed, the X–1 and D–588–I were used in combination. The turbojet-powered Skystreak, having longer flight duration, was used for the heavy work up to low transonic speeds, while the X–1 flights were at higher speeds.

Instrumentation for these flights made possible precise measurement of performance (lift and drag), stability and control, and aerodynamic loads. For the loads work, instrumentation covered measurements both by strain gauges and detailed pressure-distribution pickups. This basic pattern, supplemented by special recording devices to obtain other data, has been used with the later research aircraft.

NACA's first facilities at the Dry Lake base were entirely makeshift, consisting of wire cages in one Air Force hangar. In fact, Air Force facilities at the Base were generally inadequate. In 1948, the NACA took over a small hangar and built lean-to structures alongside for shops and offices. By the end of 1948, the complement of the NACA's Muroc Flight Test Unit had been increased to 60; the following year the group name was changed to High Speed Flight Research Station.

By early 1949 the Douglas Skyrocket, the second phase of the D–558 program, was nearing completion of contractor performance demonstrations, using only turbojet power. The airplane differed from the Phase I in two important details. It had both a Westinghouse J–34 turbojet engine and a R. M. I. rocket motor, and its wings were swept. At first the wing sweep was specified at 45°, but because so little about its effect was then known (except that it reduced lift capabilities of the wing) sweep was reduced first to 40° and finally to 35°. Automatic leading-edge slots and wing fences were provided to overcome some disadvantages of sweepback.

Later in 1949, the Skyrocket first flew with both jet and rocket power. It soon became clear that the rocket engine would be necessary, in addition to the turbojet, for ground take-off in the heavily loaded condition. Such a requirement meant there would be little rocket fuel available for high-speed flight at altitude. Consequently, the NACA recommended modification of the D–558–II for air-launching. While this work was being done, one of the Skyrockets was further modified by removing the jet engine and its fuel tanks, to provide additional rocket fuel for longer high-speed flight.

The Douglas Company in 1950 began air-launched flights, and early the next year began maximum performance flights for the Navy, using the all-rocket D–558–II. William Bridgeman, the pilot, that year exceeded 1,200 mph (Mach number of 1.87). In the summer of 1951, Col. Carl, USMC, reached 82,000 ft. altitude. It was not until November 20, 1953, however, that the maximum velocity of the D–558–II was attained. On that day, NACA pilot Scott Crossfield became the first man to reach a Mach number of 2, or 1,328 mph.

The problem of longitudinal stability (pitch-up), as experienced by airplanes with swept-back, low-aspect-ratio wings, was first met in flight with the D–558–II airplanes. The onset of pitch-up (where the airplane noses up abruptly) was not unexpected; wind-tunnel investigations had suggested this might happen. It was not until the problem could be studied in flight that its severity could be fully evaluated. In the Skyrocket, many types of wing devices were used in efforts to lessen the pitch-up severity.

The Skyrocket further was used effectively in the study of the violent lateral oscillations which may occur at high speed when the airplane is in low-angle-of-attack attitude. When Bridgeman made his flight at a Mach number of 1.87, he experienced lateral oscillations of as much as 75°. After a detailed study of the stability problems in flight, the NACA was able to dampen the lateral oscillations sufficiently to permit maximum flight to Mach number of 2.01.

One of the major problems faced by designers of supersonic fighters which can be studied in the wind tunnel as well as in flight is the decrease in directional stability that occurs as speed increases. What happens is that conventional lifting surfaces, such as the vertical tail, tend to lose their lift effectiveness with increase in Mach number. The positive directional stability of an airplane may decrease to the point where it is unacceptable. By increasing the wind-tunnel speed to and above this point, the advent and seriousness of the phenomenon as it affects a particular design can be studied in detail.

This problem was experienced first in the Skyrocket program. It has been studied with succeeding airplanes available to the NACA Station.

During 1949 and the first months of 1950, the Northrop X–4, a tailless, swept-wing airplane powered by two Westinghouse J–30 turbojet engines was undergoing contractor trials. Sponsored by the Air Force, the airplane was assigned to the NACA in mid-1950. For several years thereafter, the X–4 was flown many times to investigate problems peculiar to a tailless design. Stability difficulties encountered at higher speeds made impossible attainment of speed beyond a Mach number of 0.95. In the X–4 program, much of value was learned about the stability and control characteristics of this type of airplane. Perhaps of even greater importance was the ability to acquire trend information (hints of things to come) about dynamic stability involving the coupling of the aerodynamic and mass characteristics of an airplane.

By the beginning of 1952, when the variable-sweep wing Bell X–5 became available to the NACA, the staff of its High-Speed Flight Research Station numbered nearly 200. The X–5 was designed to permit flight investigation of difficulties resulting from the fact that although wing sweep was obviously of great utility in alleviating or delaying the large drag rise occuring near the speed of sound, its use entailed numerous problems associated with landing at low speeds. An airplane able to land and take off with its wings essentially straight, and then sweep them back 20° to 60° in flight, offered obvious advantages. The most modern airplanes being flown at the time had little more than 35° sweepback. It was felt necessary to acquire flight experience with airplanes having highly swept wings, up to 60°, and the X–5 offered the best means of accomplishing this.

Before the end of the year the X–5, powered by an Allison J–35 engine, was flying supersonically. Among other studies, it was used to investigate the change in gust response with large differences in sweep. When flown with 59° of sweep, the airplane showed considerably less response to gusts than when flown with 20° of sweep. In fact, although lateral stability deteriorated as expected with the extreme angle of sweep, pilots could fly it with 59° of sweep under atmospheric conditions that were impossible with 20°. The airplane was simple to maintain and easy to fly. The X–5 program never was spectacular but the airplane proved itself a most useful research vehicle.

In 1952 the research airplane program was broadened to include flight studies with the XF–92A. Convair built this model to acquire experience with the delta-wing configuration prior to design of the F–102. The XF–92A originally was powered by an Allison J–33 turbojet; modified to use an afterburner, it was capable of transonic performance. The NACA cooperated with the Air Force in evaluating the airplane, which work yielded some of the first full-scale information about drag of delta-wing designs. Detailed measurements were made of pressure distributions over the aircraft surfaces.

Perhaps most important among test results was the evaluation of pitch-up characteristics of the delta-wing, which proved unlike those of the other research airplanes. In flight at Edwards and in NACA wind tunnels, the XF–92A program concentrated on problems of pitch-up and lateral stability, resulting in an arrangement of wing fences which gave satisfactory flight characteristics later on the F–102.

A "by-product" of the research-airplane program was the cooperative effort in 1952 by the Air Force, Northrop, and the NACA to determine the cause of a series of structural failures in the F-89. It was believed the wings might be under strength because loadings encountered in flight were higher than stress-analysis estimates. The NACA Station had large experience in the art and science of flight measurement of aerodynamic loads by strain gauges. NACA scientists were assigned temporarily to the Edwards Northrop facility to direct installation and calibration of strain gauges, and reduction and analysis of data to determine the validity of beliefs. Once this work was done and remedies were applied, the F-89 structural failures ended.

From 1950 to 1953 the NACA Station continued to grow, but slowly, reaching 222 employees in early 1953. Additional temporary structures were erected and other temporary buildings were obtained from the Air Force. The result was an operation spread over a greater part of the east end of Edwards AFB. Meantime, beginning in 1951, the USAF master plan for enlargement of the Base was approved, and construction of large, permanent structures began. At about the same time, it became apparent the NACA flight research operation at Edwards would be continued indefinitely. A permanent facility was requested and approved by the Congress, with appropriations voted for the fiscal year 1952. The Air Force leased 175 acres of land to NACA, and in 1953 NACA began construction of a large, single building to include hanger space, instrumentation facilities, shops and offices.

In 1953, the High-Speed Flight Research Station obtained a Boeing B-47 for study of the problems peculiar to an airplane with a high degree of structural flexibility. It was desired especially to investigate aeroelastic effects on the loading encountered by this airplane in flight and its stability and control characteristics. The test program was conducted in cooperation with the Langley and Ames Aeronautical Laboratories. This long-term program, now completed, has provided valuable information about the effects of the flexing of the wing, fuselage, and tail group upon the characteristics of an airplane. At present the B–47 is being used in a program to measure accurately the noise created by a large turbojet-powered airplane and the several sources of noise. It is hoped the research results will aid efforts to quiet the jet transports which will soon be put into commercial service in the United States.

Flight tests of the Douglas X–3, powered by two Westinghouse J–34 turbojet engines, began in 1953. The Air Force sponsored airplane was delivered the following year to the NACA. This research craft, because of development difficulties with more advanced engines, never reached the desired performance goals.

Nevertheless, its layout (a very thin, straight wing mounted on a long fuselage in which most of the weight was concentrated along the centerline) generally represented the design thinking about high-performance tactical aircraft of the future. As such, the X–3 was of considerable interest, especially in the field of airplane dynamics, and much valuable information was obtained in flights reaching into the low supersonic range.

It was in the X–3 that the frightening phenomenon, now known generally as inertia coupling, was first experienced. Inertia coupling, characterized by large divergent motions of the airplane in rapid aileron rolls, was determined to result from high rates of rolling and from the concentration of the weight of the airplane within the fuselage and on a great length.

Work to analyze and understand the problem was just getting well under way in mid-1954, when an NACA aeronautical research pilot experienced the inertia coupling in a North American F–100 Super Sabre. During preliminary tests the airplane exhibited strange behavior in rapid aileron rolling maneuvers, resulting in large divergencies in yaw and pitch that could be both destructive to the airplane and, to say the least, extremely disconcerting to the pilot.

The F–100 was just going into Air Force service, where the phenomenon likewise was encountered Several airplanes were lost and the F–100 was grounded until a solution to the problem could be found. Following intensive cooperative action by the Air Force, North American Aviation, Inc., and the NACA, a joint flight test program was undertaken. Fortunately, both improved understanding of the phenomenon and a practical solution were soon determined. By enlarging the vertical tail of the airplane and performing other seemingly minor modifications, the airplane was "cured" of its troubles and soon returned to full service.

In 1954, when the new quarters for the NACA at Edwards were completed and occupied, the name of the unit was changed to High Speed Flight Station. At the same time it was raised to independent status, directly under NACA Headquarters instead of reporting to the Langley Laboratory. Additions to the staff since have increased its total complement to 300.

Research on problems of delta-wing airplanes continued with the acquisition in the fall of 1954 of the

Convair YF–102. Flight measurements of lift and drag with this aircraft were especially useful in corroborating wind-tunnel data on Richard T. Whitcomb's area rule. These data indicated that the Whitcomb principle would be valuable when applied to the Convair airplane. A study also was made of the inertia-coupling characteristics and, on the basis of information about the delta wing, together with data from the F–100 and X–3 designs, the manufacturer was enabled to free the production F–102A of inertia-coupling troubles. More recently, the High Speed Flight Station has received an F–102A. This model permits extension of flight research with delta-wing airplanes into the supersonic range.

Looking back for a moment, the X–1A, carrying a larger store of rocket fuel than the original X–1, underwent contractor trials in 1952–53. This program was climaxed December 12, 1953 when Maj. Yeager reached a Mach number of 2.5, approximately 1,650 mph. Military pilots later flew it to altitudes as high as 90,000 feet.

In 1954, both the X–1A and the X–1B (specially instrumented for full scale aerodynamic-heating research in flight) were delivered to the NACA. The first of these was lost in 1955 when, in the course of air launching from the B–29 mother ship, an explosion caused serious structural damage to the X–1A. Unable to jettison fuel, the crew found it necessary to release the research plane and let if fall to earth. Fortunately there was no injury to personnel—due in no small part to the heroism of the flight crews.

Among the newer airplanes now being used at the Station is the Lockheed XF–104A, which is capable of speeds near a Mach number of 2.0. This airplane will offer the opportunity to study inlet problems or problems of air-breathing engines, as well as problems associated with sustained high supersonic speeds, including aerodynamic heating, supersonic speed stability, and related subjects.

Another of the X–1 series, the "E" model, was modified from one of the original X–1's. It incorporates a 4-percent-thick wing and a new propellant and tankage system. The X–1E is being used to study problems of very thin wings (proportionally about as thin as a razor blade) at high supersonic speeds and altitudes.

In 1956, following more than a decade of development effort, the Bell X–2 became operational. Made of stainless steel and Monel metal, it was designed for study of aerodynamic heating at speeds near, if not above, a Mach number of 3.0. Because of fabrication difficulties inevitable with new materials and techniques, and even more, because of problems of developing satisfactory throttling apparatus for the Curtiss-Wright rocket engine, the airplane was delivered several years late.

Late in the spring, Col. Frank Everest, USAF, flew the X–2 to a new speed record and soon after, Maj. Iven Kincheloe climbed to a higher altitude than ever previously reached by man. In one of the last flights scheduled in the Air Force performance program, Capt. Milburn G. Apt was killed.

The tragic loss of the Bell X–2 inevitably will slow the rate of full-scale research on aerodynamic-heating problems, but both the X–1B and X–1E are being used profitably in this program. The first of these airplanes, instrumented with some 500 thermocouples to permit detailed evaluation of the heating problem, can be flown at speeds above a Mach number of 2.0. The thin-wing X–1E, with its increased fuel supply, can fly above a Mach number of 2.0 for a somewhat longer time.

How serious the aerodynamic-heating problem already has become may be seen from the fact that at sustained speed in the Mach number 2.0 range a temperature rise of as much as 300° F may be experienced. Further, the temperature due to aerodynamic heating rises as the square of the velocity: at a Mach number of 3.0 the temperature rise during sustained flight would be about 660° F.

Also planned for use in study of such problems as aerodynamic heating and stability and control at high altitude, and now under construction, is the X–15. North American Aviation, Inc., is the contractor, and the Air Force the sponsoring service. As with previous research airplanes, the NACA is an active partner, participating fully in the grave decisions implicit in design and construction of an airplane of new concept that is intended to fly still faster and higher than man has attempted. Because of the loss of the X–2, the need for earliest possible completion of this project has become imperative.

AERODYNAMIC RESEARCH

Man's efforts to fly faster and higher have never before been so intense, and success is dependent upon gaining a greater knowledge and understanding of the fundamental problems of fluid mechanics of supersonic and hypersonic flows along with other factors affecting aircraft performance, stability, and control. Hence a major portion of NACA research has been devoted to experimental and theoretical aerodynamic investigations. In addition to many studies of generalized aircraft configurations, programs have been undertaken at the request of the military services to assist in the development of specific military airplanes or missiles. Such studies uncover problems which lead to general research programs. In the past year, for example, considerable effort has been expended on the methods of eliminating the decrease in directional stability associated with increasing Mach number and angle of attack, which is a serious problem in the design of high-speed airplanes.

Aerodynamic problems of the long-range ballistic missile, such as aerodynamic heating, dynamic stability, and the determination of optimum shapes for reentering the earth's atmosphere, have also become increasingly important.

The results obtained from flight programs on research airplanes undertaken at the NACA High-Speed Flight Station are of considerable value for future airplane designs. Flight programs conducted on current service airplanes by the various laboratory flight groups are also of special value to the services and their contractors. The results often lead to the development of more advanced airplanes and, in particular, aid in obtaining satisfactory flying qualities.

Continued assistance has been given the NACA in planning and conducting its aerodynamic research programs by the Committee on Aerodynamics and its technical Subcommittees on Fluid Mechanics, High-Speed Aerodynamics, Internal Flow, Propellers for Aircraft, Seaplanes, and Helicopters. In order to handle the large field formerly covered by the Subcommittee on Stability and Control more effectively, two new subcommittees were formed during the year: The Subcommittee on Aerodynamic Stability and Control and the Subcommittee on Automatic Stabilization and Control.

As in previous years, dissemination of important new aerodynamic research results by means of a special NACA conference was found to be effective. The conference, attended by representatives of the military services and many of their contractors, was concerned with the aerodynamic problems of high-speed aircraft.

Some of the recent unclassified studies conducted by the NACA in the field of aerodynamics are described in the following paragraphs.

FLUID MECHANICS

Boundary-Layer Research

The importance of obtaining extensive regions of laminar flow on all surfaces during high-speed flight stems from the benefits of reduced heat transfer, equilibrium temperatures, and skin friction. The Reynolds number for transition at a Mach number of 6.9 was found in the Langley 11-inch hypersonic tunnel. The test model was a hollow cylinder and measurements were made on its outside surface with heat transfer from the boundary layer to the wall. The boundary-layer nature was determined from impact-pressure surveys. As reported in Technical Note 3546, the data obtained at a Reynolds number of 0.34×10^6 per inch in a portion of the nozzle which had a small Mach number variation show that the transition Reynolds number occurred between 4×10^6 and 6×10^6. When the cylinder protruded into a region of the nozzle where a considerable negative pressure gradient existed, the Reynolds number for transition approached 8×10^6 for one set of data. At a Reynolds number of 0.26×10^6 per inch, the Reynolds number for transition varied from about 4×10^6 to 4.5×10^6. From a correlation of results obtained at lower supersonic Mach numbers (Mach numbers from 2.0 to 4.5), leading-edge thickness and free-stream Reynolds number per inch appear important in determining flat-plate transition. Results from various research facilities would not appear to be comparable unless these factors are taken into account. At a given Mach number the Reynolds number based on leading-edge thickness appears to be a significant parameter and should be considered in comparisons of flat-plate or hollow-cylinder boundary layer-transition data obtained from various research facilities.

A number of the factors that presumably affect boundary-layer transition have been investigated at the Lewis Laboratory. The effects on transition of heating and cooling the surfaces of cone-cylinder and parabolic-nose-cylinder models are reported in Technical Note 3562. Cooling the cone-cylinder model to a ratio of wall to free-stream static temperature of approximately 1.4 increased the transition Reynolds number from about 2.0×10^6 to 10.6×10^6 at equilibrium. For temperature ratios less than 1.4, the boundary-layer flow was laminar over the entire model.

For the parabolic-nosed body, the boundary-layer-transition Reynolds number was about twice that of the cone-cylinder model over the temperature range investigated.

Another factor considered was the effect of leading-edge geometry on transition position, recovery-factor distribution, boundary-layer profile, and the roughness required to induce transition. Results of the investigation conducted at Mach 3.1 with a hollow cylinder alined with the air stream are reported in Technical Note 3659. A large downstream displacement of the transition point and an increase in recovery factor were noted when a sharp leading edge was blunted very slightly. These effects may be attributed to the formation of an inviscid shear layer near the surface which was caused by the curvature of the leading-edge shock. The boundary layer thus develops in a region of lower Mach number which exists within this shock-produced shear layer. The delay in transition is predominantly an effect of a Reynolds number reduction within the reduced velocity region of the inviscid shear layer. A still larger downstream displacement of the transition point was observed for an externally beveled leading edge.

Much work has been done to determine the transition point on bodies of revolution and unswept surfaces at zero angle of attack. For supersonic configurations, however swept wings operating at angles of attack are common, and questions about the effects of sweep and angle of attack on boundary-layer transition arise. Very little theoretical or experimental work has been done on these problems at supersonic speeds.

An experimental investigation was therefore conducted at the Langley Laboratory to determine the effects of leading-edge sweep, angle of attack, and leading-edge thickness on boundary-layer transition of flat-plate wings at a Mach number of 4.04. The data presented in Technical Note 3473 show that transition always occurred along a front parallel to the leading edge of the wing. Increasing the leading-edge sweep angle or increasing the angle of attack between the undisturbed air stream and the wing surface caused the transition line to move closer to the wing leading edge and generally decreased the Reynolds number at which transition occurred. An increase in leading-edge thickness from 0.25 mil to 6 mils caused large increases in the local Reynolds numbers at which transition occurs for a flat-plate wing with an unswept leading edge. On wings with 45° and 60° leading-edge sweep, however, an increase in the leading-edge thickness had no apparent effect on the local normal transition Reynolds number. For small angles of leading-edge sweep, the favorable pressure gradient due to the curved profile of the NACA 65A004 airfoil section produced longer lengths of laminar flow than were obtained on the flat-plate section. For larger sweep angles, the destabilizing effect of the curved streamline outside the boundary layer caused transition to occur earlier than on the flat plate.

Until recently, studies of unsteady laminar boundary layers were limited to either the early stages of the motion (i. e., the transient state) or oscillatory motions without a mean flow. The fluid, furthermore, was assumed to be incompressible. It was felt that the boundary-layer growth occurred in so short a period of time that, for engineering purposes, the flow could be assumed to be steady. However, in many present-day applications, consideration must be given to the unsteady flow effects for long time periods and to high-speed flows in which compressibility becomes important. For example, the skin friction and heat transfer of a rocket missile varies over its entire flight because of the unsteady flow caused by the varying flight speed throughout its trajectory. Blades rotating in nonuniform air streams, unsteady nozzle flow, and oscillating wings are some of the other important cases.

The results of a study at the Lewis Laboratory concerned with such unsteady flow are given in Technical Note 3569. In this paper, consideration is given to the laminar compressible boundary layer and heat transfer over a semi-infinite flat plate. The plate was maintained at a uniform (both temporally and spatially) temperature and moved with a continuous but otherwise arbitrary time-dependent velocity. The solutions were obtained as a series about the quasi-steady state.

The lack of knowledge about the proper definition of an "origin," that is, of an absolute streamwise Reynolds number scale of a turbulent boundary layer, makes it difficult to find a satisfactory way of comparing skin-friction coefficients of low- and high-speed turbulent boundary layers.

This problem, which is of major importance at some Reynolds numbers, is discussed in Technical Note 3486. The report describes the design and construction of a floating-element skin-friction balance. This instrument, which is similar to Dhawan's balance, measured the local skin friction in the turbulent boundary layer of a smooth flat plate at high-subsonic Mach numbers and supersonic Mach numbers up to 1.75. The measured skin-friction coefficients are consistent with the results of other investigations at subsonic and supersonic speeds. The principal difficulties in comparing skin-friction coefficients at various Mach numbers are discussed. The study was conducted by the Guggenheim Aeronautical Laboratory of the California Institute of Technology under the sponsorship of the NACA.

In recent years the phenomena associated with shock-induced separation of the boundary layer have received increased attention because of the important influence that separation may have upon the overall aerodynamic characteristics of complete aircraft configurations by affecting the performance of individual components. A study has been made at the Langley Laboratory of some recent contributions to the problem of shock-induced boundary-layer separation. Analytical and experimental results of this study are presented in Technical Note 3601. The probable ranges within which the pressure rise and flow deflections associated with separation may be expected to lie are shown. Consideration is given to the effects of Mach number, adverse pressure gradients, and Reynolds number for laminar boundary layers, and to the effects of Mach number, Reynolds number, and ratio of specific heat, for turbulent boundary layers.

The results of a study at the Lewis Laboratory of the boundary layer behind a shock wave advancing into a stationary fluid are presented in Technical Note 3712. In this study, the laminar-boundary-layer problem, except for the weak wave case (which can be solved analytically), was solved by numerical integration. Integral (Kármán-Pohlhausen type) solutions were obtained to provide a guide for determining expressions which accurately represent the numerical data. Analytical expressions for various boundary-layer parameters are presented and these agree with the numerical integrations to within 1 percent. The turbulent-boundary-layer problem was solved by using integral methods similar to those employed for the solution of turbulent compressible flow over a semi-infinite flat plate. The fluid velocity, relative to the wall, was assumed to have a seventh-power profile. A form of the Blasius equation which accounted for compressibility and related turbulent skin friction and boundary-layer thickness was utilized.

In flows which are suddenly accelerated in a shock tube, departures of the real fluid from results predicted by both the nonviscous-unsteady-flow theory and the viscous steady-flow theory occur. The effect of skin friction has been studied at the Langley Laboratory by artificially increasing the friction by using surface roughness, which created very thick boundary layers. Measurements of the unsteady-flow turbulent-boundary-layer characteristics were made using a new technique involving study of the shape of the bow wave from a bullet fired through the flow. The boundary-layer-thickness measurements agreed well with both steady-flow experiments and theory. The measured unsteady-flow velocity profiles were considerably less full than the steady-flow theoretical profiles but agreed with some of the steady-flow data. Shock-attenuation measurements were also made and agreed with the results of simple theory at small values of boundary-layer-displacement thickness but were much lower than theoretical results at large values. Results of this study are presented in Technical Note 3627.

Heat Transfer

Aerodynamic heating in supersonic flight has long been recognized as a major problem in the design of supersonic aircraft, and experimental heat-transfer data for high Mach numbers and Reynolds numbers are in great demand. Except for some work done on the V–2 rocket, most of the convective heat-transfer work has been done in wind tunnels under steady-state conditions. In Technical Note 3623 results are presented for the transient conditions encountered along the trajectory of two parabolic bodies of revolution as obtained in free flight at the Langley Pilotless Aircraft Research Division. The tests covered Mach numbers from 1.02 to 2.48 and Reynolds numbers from 3×10^6 to 164×10^6. The experimental heat-transfer coefficient values obtained are about 15 percent higher than those measured in other tests on a V–2 research missile and approximate the values given by subsonic flat-plate theory with a Reynolds number analogy factor of 0.6.

In order to extend the Mach number range of previous studies of heat transfer over hemisphere cylinders, an investigation was conducted in the Langley 11-inch hypersonic tunnel at a Mach number of 6.8 and Reynolds numbers from approximately 1.09×10^5 to 1.03×10^6 based on diameter and free-stream conditions. In this investigation the boundary layer was laminar and had average T_s/T_∞ values of about 7 at the nose and about 6 on the cylinder (T_s is the local free-stream temperature just outside the boundary layer and T_∞ is the free-stream temperature ahead of the normal shock). The investigation utilized data from transient nonisothermal temperature distributions since, at the high-temperature levels involved, an isothermal surface is difficult to obtain. The data presented in Technical Note 3706 have been correlated with theoretical analyses and the results of other experiments obtained at lower Mach numbers. The experimental heat-transfer coefficients from this investigation were slightly less over the whole body than those predicted by the theory of Stine and Wanlass for an isothermal surface. A modification of Sibulkin's stagnation-point solution gave the trend of the local Stanton number with local Reynolds number for angles up to 45° from the stagnation point. The calculated values, however, were approximately 12 percent higher than the experimental values.

Pitot profiles taken in this study at a Mach number of 6.8 verify that the local Mach number or velocity outside the boundary layer, required in the use of theory, may be computed from the surface pressures by using isentropic flow relations and conditions behind a normal shock. The experimental pressure distribution at a Mach number of 6.8 is closely predicted by the modified Newtonian theory. The velocity gradients calculated by using the modified Newtonian theory at the stagnation point vary with Mach number and are in good agreement with those obtained from measured pressures for Mach numbers from 1.2 to 6.8.

At the stagnation point a second modification of the theory of Sibulkin using the diameter and condition behind the normal shock was in good agreement with experiment when the velocity gradient at the stagnation point appropriate to the free-stream Mach number was used.

Engineering information relative to the heat-transfer coefficient and temperature recovery factor is required over a wide range of Mach numbers and Reynolds numbers. Such information is especially important in the computation of aerodynamic heating. A project is under way at the Langley Laboratory to obtain such information for the zero-pressure-gradient, two-dimensional case with turbulent flow for values of Reynolds number from about 1×10^6 to 1×10^8. Previously, data have been published at Mach numbers of 1.62, 2.06, and 3.03. In the past year the range has been extended with results for a Mach number of 0.87 published in Technical Note 3599. The heat-transfer coefficients and recovery factors determined in this study are in fairly good agreement with the theoretical predictions of Van Driest throughout the investigated range of Mach numbers and Reynolds numbers.

An interesting natural convection problem is that concerning the thermally unstable configuration obtained when a fluid is heated from below. The effects of frictional heating and heat sources on this phenomenon, as determined from a study at the Lewis Laboratory, are reported in Technical Note 3458. In this study, the effects of heat sources and frictional heating on the laminar fully developed channel flow subject to a body force between two parallel plates oriented in the direction of the body force are analyzed. Solutions are obtained for combined forced- and natural-convection flows for the cases in which the wall temperature variations are linear and (1) the wall temperatures are specified, (2) the walls are insulated, and (3) the net mass flow in the channel is zero. These solutions depend on the Rayleigh number which was previously found to be the factor determining the stability and type of flow for horizontal and vertical layers of fluids heated from below but without heat sources or frictional heating. Similar stability characteristics were displayed in this study, and the heat sources affect the flows only in a quantitative manner.

The cooling of aerodynamically heated surfaces has gained attention in connection with rocket walls, turbine blades, and high-speed flow over aircraft surfaces. A promising means of cooling such surfaces appears to be transpiration or sweat cooling, a method in which the surface is made porous and a comparatively small quantity of cool fluid per unit time is injected through the pores into the main stream. Separation may be of particular interest in connection with flow over a transpiration-cooled surface because a normal mass flow strongly tends to promote separation by moving the separation point upstream. On the other hand, cooling of the wall tends, by itself, to delay separation by moving the separation point downstream.

In order to determine the actual net effect of simultaneous normal mass flow and cooling of the wall on conditions of separation over a sweat-cooled surface, a theoretical analysis of laminar separation in compressible flow over a transpiration-cooled surface maintained at a uniform wall temperature has been made at the Polytechnic Institute of Brooklyn under NACA sponsorship. In Technical Note 3559 a simple method of calculating the separation point over such surfaces for a given adverse pressure gradient, Mach number, wall temperature, and uniform coolant temperature is developed. This method is expected to be sufficiently accurate for most purposes. To show the effects of these parameters on the separation point a numerical example is worked out in detail. The normal mass flow was found to have a predominate effect on the separation-point location, since separation over a transpiration-cooled wall occurs upstream of the separation point location of a heat-insulated wall without normal mass flow at the same adverse pressure gradient and Mach number.

Gas Dynamics

The calculation of flows about objects, primarily missiles, traveling at high supersonic speeds is a matter of great practical interest. These calculations are difficult because at high supersonic speeds the disturbance velocities are not necessarily small compared with the velocity of sound, nor are entropy gradients necessarily negligible in the disturbed flow field about a body, even though it may be of normal slenderness. Thus, for example, the simple linear theory, which is valuable in studying flows at low supersonic speeds, loses much of its utility in the study of high-supersonic-speed flows.

A procedure has been developed at the Ames Laboratory for calculating three-dimensional steady and nonsteady supersonic flows using the method of characteristics. An approximate method was deduced and shown to be of practical value for pointed bodies of revolution at high supersonic speeds. In the application of the approximate method, flow at the vertex is analyzed using a generalized Prandtl-Meyer expansion theory. Surface pressures and bow shock waves for inclined ogives determined by using the approximate method were compared with experimental results obtained at Mach numbers from 3 to 6.3. Theoretical predictions and experimental results are in good agreement for values of the hypersonic similarity parameter greater than about 1. It was also found that the concept of two-dimensionality in inviscid hypersonic flows has a counterpart in hypersonic boundary-layer flows. From this result, a unified two-dimensional approach to three-dimensional hypersonic flows was developed. The results of this investigation are presented in Report 1249.

The application of this theory to bodies traveling at large Mach numbers is often limited, however, by the restriction that the maximum slope of the body must be less than the slope of a free-stream Mach line. The applicability of the generalized shock-expansion method and that of the second-order potential theory do not always overlap, and there remain flows at certain combinations of Mach number and body shape which cannot be treated by either theory. Normally, these intermediate flows are encountered when the hypersonic similarity parameter based on nose fineness ratio is in the neighborhood of 1.

Since this is a range of practical interest, a new second-order shock-expansion method has been developed at the Ames Laboratory. This method applies to flows about bodies of revolution characterized by values of the hypersonic similarity parameter near 1 (i. e., the ratio of free-stream Mach number to body fineness ratio). The method has been applied to the calculation of pressure distribution, normal-force curve slope, and center-of-pressure position on bodies of revolution at zero lift. An experimental investigation was conducted in the Ames 10- by 14-inch supersonic wind tunnel to assess the accuracy of the method. Cone-cylinder and ogive-cylinder bodies with fineness ratios from 3 to 17 were tested at Mach numbers from 3.0 to 6.3 to determine normal-force-curve slope and center-of-pressure position at zero lift. The theoretical method predicts these aerodynamic characteristics within the accuracy of the experimental results throughout the range of test variables investigated. The results of this investigation are reported in Technical Note 3527.

A method based upon the theory of characteristics has been developed at the Langley Laboratory to compute the contour of a body of revolution for a prescribed pressure distribution at isupersonc speeds. The design of such a body would be useful in the investigation of many aerodynamic problems such as those associated with body-wing interaction, inlets, boundary-layer transition, separation, and shock-wave interaction. The evaluation of the flow properties at any point along the surface of the body is determined by an iteration process. This method, described in Technical Note 3555, was used to compute the contour of a body with a constant pressure gradient. This pressure distribution was chosen because the theoretical approach to the transition problem indicated that a constant pressure gradient would be desirable for certain experimental studies. A model was constructed and tested at Mach numbers of 3.05 and 3.13. Excellent agreement with the computed distribution was obtained.

Even when the compressible-flow equations can be solved precisely in a particular case, the numerical computation is usually laborious. If it becomes necessary to extend the calculations to related and perhaps more complicated cases, it is often desirable, or necessary to resort to approximate solutions. In Technical Note 3485, it is shown that the streamlines near an unyawed circular cone with an attached shock wave are, to a first approximation, portions of hyperbolas. This result is used as a basis for the development of an approximate solution for the cone in which the shock-wave orientation and the flow field behind the shock wave are given explicitly in terms of the free-stream Mach number, the vertex angle of the body cone, and the ratio of specific heats of the gas. The possible application of a part of this solution to the analysis of real-gas effects on the high-speed flow over a cone is discussed.

One of the transonic flow problems of interest is that of a double-wedge airfoil in slightly supersonic flow. Previous experimental work has provided a detailed description of the flow characteristics for nonlifting wedges. The data for lifting wedges, however, are far less complete. Measurements were, therefore, made at the Langley Gas Dynamics Laboratory of the flow around two wedges at Mach numbers of 1.30 and 1.41 by means of a Mach-Zehnder interferometer for various angles of attack up to 5°. The results presented in Technical Note 3626 show that pressure distributions for different Mach numbers on wedges of different thickness are similar at the same values of transonic similarity parameter and reduced angle of attack for angles of attack as large as the thickness ratio, that the lift-curve slope is nearly independent of the angle of attack for angles of attack from −2° to 2°, and that, for the airfoils tested at Mach numbers greater than the attachment value, the center-of-pressure location is nearly independent of the angle of attack.

The problem of determining the shape of slender boattail bodies of revolution for minimum wave drag has been reexamined. In Technical Note 3478 it is shown that minimum solutions for Ward's slender-body drag equation can exist only for the restricted class of bodies for which the rate of change of cross-sectional area at the base is zero. In order to eliminate this restriction, certain higher order terms must be retained in the drag equation and isoperimetric relations. The minimum problem for the isoperimetric conditions of given length, volume, and base area is treated as an example. According to Ward's drag equation, the resulting body shapes have slightly less drag than those determined by previous investigators.

If especially advantageous arrangements for supersonic aircraft are not to be overlooked it becomes necessary to study completely general or arbitrary arrangements of airfoils and bodies in three-dimensional space. In Technical Note 3530 the problem of minimizing the wave drag for a given total volume and a given total lift is discussed. The volume and lift may be carried by any number of slender bodies or thin airfoils, but to make the problem definite, they are confined to the interior of a given fixed region which may outline the maximum permissible dimensions of the aircraft.

In a related study, reported in Technical Note 3667, several variations involving optimum wing and body combinations having minimum wave drags are analyzed for different geometrical restraints. Particular attention is paid to the effect on the wave drag of shortening the fuselage and, for slender axially symmetric bodies, the effect of fixing the fuselage diameter at several points or even of fixing portions of its shape.

A great deal of effort is presently being expended in correlating the zero-lift drag rise of wing-body combinations on the basis of their streamwise distribution of cross-section area. This work is based on the discovery of the concept, known as the transonic area rule, that "near the speed of sound, the zero-lift drag rise of thin low-aspect-ratio wing-body combinations is primarily dependent on the axial distribution of cross-sectional area normal to the airstream." Since an accurate prediction of drag is vital to the designer and since the use of such a simple rule is appealing, it is very important to investigate the applicability of the rule to the widest variety of shapes of aerodynamic interest. Some insight into the applicable range of the transonic area rule has been gained in a study at the Ames Laboratory comparing it with the appropriate similarity rule of transonic-flow theory and its results with available experimental data for a large family of rectangular wings having NACA 63AXXX profiles. In spite of the small

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number of geometric variables available for such a family, the range is sufficient to include cases both compatible and incompatible with the area rule. A résumé of the study is given in Technical Note 3673.

The precise experimental determination of the structure and thickness of the normal shock wave is of fundamental interest in the study of gas dynamics because it does much to define the usefulness of the Navier-Stokes equations, or of alternative equations, for predicting the behavior of a nonuniform gas. In addition, observations of the nature of the shock wave shed considerable light on the magnitude and character of so-called "relaxation effects" associated with the finite time required to obtain equipartition of energy among the translational and internal motions of a polyatomic molecule.

To provide additional information, the profiles and thicknesses of normal shock waves of moderate strength have been determined experimentally at the University of California under the sponsorship of the NACA in terms of the variation of the equilibrium temperature of an insulated transverse cylinder in free-molecule flow. Steady-state shock waves were produced in the jet of a low-density wind tunnel at initial Mach numbers of 1.72 and 1.82 in helium and 1.78, 1.85, 1.90, 1.98, 3.70, and 3.91 in air. The shock thickness, determined from the maximum slope of the cylinder temperature profile, varied from 3½ to 5 times the length of the Maxwell mean free path in the supersonic stream. A comparison between the experimental shock profiles and various theoretical predictions leads to the tentative conclusions that: (1) The Navier-Stokes equations are adequate for the description of the shock transition for initial Mach numbers up to 2, and (2) the effects of rotational relaxation times in air can be accounted for by the introduction of a "second" or "bulk" viscosity coefficient equal to about two-thirds of the ordinary shear viscosity.

The results of a study conducted at Iowa State College under the sponsorship of the NACA in which the relaxation times for the excitation of molecular vibrations in a number of heavy gases were measured with an acoustic interferometer are reported in Technical Note 3558. All the gases studied were found to have a single relaxation time, indicating that intermodal coupling in these gases is strong. Only binary collisions were found to be important in the excitation of vibrations. The probability, in a given collision, of exciting or deexciting molecular vibrations in the halogen-substituted methanes appears to depend upon the relative energy of approach of the colliding molecules rather than upon the relative velocity. The excitation of vibrations in polyatomic gases may involve the formation of a complex molecule with a very short life.

The shock waves produced during flight at higher hypersonic speeds result in air temperatures high enough to cause at least a partial dissociation of the air molecules. This dissociation has some effect on the rate of aerodynamic heating. Since dissociation occurs at a finite rate, a thorough understanding of aerodynamic heating requires information on the rate of dissociation. In Technical Note 3634, the collision-theory equation for rate of dissociation has been applied to the flow behind normal shock waves at Mach numbers of 10, 12, and 14 to find the distance required for a moderately large fraction of the oxygen to dissociate. This distance was found to vary widely with Mach number and density from a fraction of a millimeter to hundreds of meters.

Research Equipment and Techniques

Proper design of wings, flaps, inlets, or wind tunnels in which area suction is to be used requires accurate and comprehensive information on the permeability characteristics of the porous materials involved. In most instances, the available information on each porous material was obtained with a single fixed relation between Mach number and Reynolds number, which is not sufficient for predicting permeability characteristics for a range of operating conditions. Experiments performed in the Langley 19-foot pressure tunnel on rolled wire cloth and sintered bronze showed that the permeability characteristics were affected by absolute pressure, flow choking, and material thickness. This information has been presented in Technical Note 3596 together with simple calculation and correlation procedures for determining permeability characteristics with reasonable accuracy when experimental data are limited.

A general investigation is being conducted to determine the relative smoothness of the flows in the various supersonic facilities of the National Advisory Committee for Aeronautics. As part of this investigation, tests have been conducted in the Langley 4- by 4-foot supersonic wind tunnel to determine the transition Reynolds number for a 10° cone at Mach numbers of 1.41, 1.61, and 2.01. The results of these tests, given in Technical Note 3648, indicate that, on the average, the transition Reynolds numbers for a smooth cone increased with tunnel stagnation pressure from about 7×10^6 at a test Reynolds number of 4×10^6 per foot to approximately 8×10^6 at a test Reynolds number of 9×10^6 per foot for all test Mach numbers. There was no effect of Mach number on transition Reynolds number in the range investigated. The results also indicated that the transition point was unsteady and tended to oscillate approximately ±10 percent about the mean value of transition Reynolds number. The values of transition Reynolds number obtained were higher than those for any other tunnel for which similar data were available, which is indicative of a low tunnel turbulence level and freedom from extraneous flow distrubances. In addition, it was found that a single-element two-dimensional surface roughness of one layer of ½-inch-wide and 0.003-inch-thick cellulose tape caused a larger decrease in transition Reynolds number than was experienced in low-speed or in other supersonic wind-tunnel investigations.

A continuing search is being carried out for techniques for producing high-stagnation-temperature hypersonic airstreams. One method involves use of the heat release given by chemical reactions. Here, the requirement is that not only the chemical used must liberate sufficient heat but also that the products of the reaction must have as nearly as possible the same flow properties as air for tunnel flow. Nitrous oxide was found to be promising for this use. Theory indicates that the thermal decomposition of nitrous oxide at constant volume can produce a temperature of 4,050° F and that the end products consist of two parts of nitrogen to one part of oxygen, which is reasonably close to the composition of air.

Preliminary laboratory tests made to verify theory for both a constant-volume and a constant-pressure decomposition show, as reported in Technical Note 3624, that a more complete decomposition was obtained at high-pressure conditions. Pressure up to 1,930 atmospheres was obtained for the constant-volume process, whereas the constant-pressure process was operated at 70 atmospheres, and theoretically predicted temperatures were approached. Decompositions up to 98 percent complete for the constant-volume process and 90 percent complete for the constant-pressure process were obtained. Results of comparison tests made in a small-scale wind tunnel at Mach numbers up to about 7 on air and on the products of nitrous oxide decomposed at constant pressure showed no significant difference.

In order to simplify the problem of model support for the measurement of forces at large angles of attack at high Reynolds numbers and supersonic speeds, nozzles exhausting directly to the atmosphere are used at the Langley gas dynamics laboratory. At the higher Mach numbers the separation of the boundary layer from the nozzle contours resulted in excessive stagnation pressures being required for establishing supersonic flow in the test section. An investigation, reported in Technical Note 3545, was made at Mach numbers of 2.7, 3.0, 3.5, 4.0, and 4.5 to determine the effect of short, fixed, wedge diffusers on the starting characteristics of these jets. Wedge diffusers that were extensions of the nozzle contours reduced the pressure ratios required for starting to less than one-half the values obtained without a diffuser. Central-body diffusers were not so effective as wedge extensions of the nozzle contours. With the wedge extensions of the nozzle contours, the jets could be started at each test Mach number for values of diffuser minimum area considerably below the values predicted by one-dimensional theory. The central-body diffusers required diffuser minimum area at least as large as theoretical values.

A series of wind-tunnel investigations has been conducted to determine the effect of inclination of the airstream on the measured pressures of a number of total-pressure tubes through a wide angle-of-attack range at subsonic, transonic, and supersonic speeds.

These investigations were conducted to obtain information which would lead to the design of rigid-type tubes capable of measuring total pressures correctly at high angles of attack and at high speeds. The need for this information has arisen because of the development of airplanes capable of maneuvering to high angles of attack at supersonic speeds and because conventional tubes, both rigid and swiveling, are unsatisfactory under these conditions. Conventional rigid tubes, that is, those with hemispherical or ogival nose shapes, are unsatisfactory because the measured pressure begins to deviate from the free-stream value at moderate angles of attack. Swiveling tubes, on the other hand, are considered undesirable because of possible structural failure at high speeds. Technical Note 3641 summarizes these results and presents the data in a form permitting a more detailed comparison of the effects of pertinent design variables.

The various tubes studied differed in regard to external shape, internal shape, and type of total-pressure entry. For simple nonshielded tubes, the best combination of these design variables produced a usable angle-of-attack range of $\pm 28°$ at a Mach number of 0.26. For the more complicated shielded tubes, the best design produced a usable angle-of-attack range of $\pm 63°$ at a Mach number of 0.26. The throats of the shielded tubes were vented through the walls of the shields, a design feature permitting end-mounting of the tube on a horizontal boom. There was considerable effect of Mach number on both unshielded and shielded tubes. For most of the unshielded tubes, the usable range increased the Mach number, whereas that of the shielded tubes decreased with Mach number.

In the description of random fields, the correlation function is an important tool. An instrument of fairly simple design for measuring time correlation functions of two stationary random electrical signals is discussed in Technical Note 3682. It is intended primarily for use in problems connected with aerodynamically produced acoustic fields but has properties suitable for application to a rather wide range of aerodynamic problems involving turbulent-flow fields. It has been designed and constructed with a view to economy and simplicity of operation and makes extensive use of the general statistical properties of the problems for which it is intended. A few experimentally determined autocorrelation functions are given in the Technical Note to indicate the degree of accuracy achieved. The Fourier transform of the autocorrelation function of a random input is compared with the power spectrum of the same function. The study was conducted at the Guggenheim Aeronautical Laboratory of the California

Institute of Technology under the sponsorship of the NACA.

HIGH-SPEED AERODYNAMICS

Airfoils and Wings

Designers of aircraft and aircraft propellers have repeatedly expressed the need for airfoil-section data in the transonic speed range. Almost all airfoil-section data in the subsonic speed range have been obtained from closed-throat tunnels which limited the tests to Mach numbers less than about 0.9. One method of extending speed range of two-dimensional experimental tests is by utilization of the open-jet principle to eliminate the choking limitations of closed-throat tunnels. Tests of a group of related NACA airfoil sections varying in maximum-thickness location, design lift coefficient, and thickness distribution have been conducted in a two-dimensional, open-throat-type wind tunnel at Mach numbers from 0.3 to about 1.0 and Reynolds numbers from 0.7×10^6 to 1.6×10^6. The results of the tests, reported in Technical Note 3607, indicate that near sonic speeds the maximum normal-force—drag ratio approaches the low values theoretically determined for a biconvex airfoil in supersonic flow. Contrary to low-speed results, the maximum normal-force—drag ratio increased as either the thickness ratio or the camber was decreased. At all Mach numbers the normal-force coefficient for maximum normal-force—drag ratio generally increased with increases in thickness ratio and camber, and also with forward movement of the position of maximum thickness. The trends of the data in the highest test Mach number range indicated that the normal-force-curve slopes of all airfoils tested are approximately equal at a Mach number of 1.0, the value being about the same as at low speeds.

As part of the NACA program to determine the zero-lift drag characteristics of various wings at supersonic, transonic, and high-subsonic speeds, the Langley Pilotless Aircraft Research Station has conducted tests of a series of 60° delta wings. These wings had varying airfoil sections and were mounted on rocket-propelled test bodies.

The sections investigated were three double-wedge airfoils of 6-percent-thickness ratio with the position of maximum thickness at 20, 50, and 80 percent of the chord, an NACA 65–006 airfoil section, and a double-wedge 3-percent-thick wing with maximum thickness at 50 percent chord. The results, determined at Mach numbers from 0.7 to 1.6 and presented in Technical Note 3650, show that the forward locations of wing maximum thickness were better from a drag standpoint and that the thinnest section gave the lowest drag. Theoretical wing drag compared well with experimental. In related tests on 6-percent-thick, 2.7-aspect-ratio rectangular wings, varying the section from a circular arc to a symmetrical diamond section, as reported in Technical Note 3548, decreased the drag at supersonic speeds and increased it at high subsonic speeds.

The aerodynamic properties of wedges of infinite span at Mach numbers near shock attachment have now been well explored. Information on wedges of finite span is less extensive. Since three-dimensional problems are beyond the reach of presently used transonic theory, knowledge here must come from experiment. A wind-tunnel study has therefore been made in the Ames 6- by 6-foot supersonic tunnel of the transonic flow over two rectangular wings of double-wedge section and aspect ratios of 2 and 4. The data cover the Mach number range from 1.166 to 1.377, which brackets the theoretical value at which the shock wave attaches to the leading edge of these wings at zero angle of attack. The effects of finite aspect ratio on the section characteristics of a double wedge were determined by comparing the present results with previous data for an infinite aspect-ratio wedge and also with two-dimensional theory for this shape.

The pressure-drag coefficient at zero lift is found to decrease with decrease in aspect ratio for all values of the test Mach number. Also, decreasing the aspect ratio reduces the lift-curve slope. The drag due to angle of attack is affected by variation of chord forces as well as normal force.

Theoretical considerations also lead to certain conclusions regarding wave detachment. In particular, detachment of a shock wave from a wedge of finite span occurred at the same free-stream Mach number as that from a wedge of infinite span. Results of this investigation are presented in Technical Note 3522.

Recent systematic experimental investigations of the effects of wing aspect ratio, thickness, and camber for wings of rectangular plan form at transonic speeds have provided data ideally suited for correlation by means of the transonic similarity parameters. Report 1253 indicates that the experimental data obtained on a series of 40 wings in the Ames 14-foot wind tunnel could be, for the most part, successfully correlated throughout the subsonic, transonic, and moderate supersonic regimes. By proper choice of parameters, the force and moment data could be presented in a concise manner effectively displaying the transonic characteristics of wings of both large and small aspect ratios. In many instances, it was found possible to predict from the correlation studies an expected range of validity for slender-body concepts. It appears that slender-body theory is adequate at sonic speed for rectangular wings of symmetrical profile if the product of the aspect ratio and the ⅓ power of the thickness ratio is less than unity. A study of simple flow compression or expansion at near sonic speeds is also included. Transonic approximations for the classical shock polar and for Prandtl-Meyer flow are derived and a limit for linearized two-dimensional flow theory at slightly supersonic speeds is given.

The problem of minimizing the supersonic drag for a given lift on a fixed plan form has been approached in several different ways. In Technical Note 3533, Lagrange's method of undetermined multipliers, is applied to the problem. The method indicates that a constant interference drag exists between the optimum loading and any other loading at the same lift coefficient. This is an integral form of a criterion established by Dr. Robert T. Jones. The best combination of four simple lift loadings on a delta wing with subsonic leading edges is calculated as a numerical example. The calculations show that the best combination of the four nonsingular loadings has about the same drag as a flat plate with full leading-edge thrust.

Recent experiments indicate that airfoil sections having sharp leading and trailing edges, heretofore considered good at supersonic speeds, may be inferior to blunt-trailing-edge airfoils when compared on the basis of drag-stiffness ratio and when used as control surfaces. The results of a Langley free-flight rocket-propelled model investigation of the effect of trailing-edge thickness on the zero-lift drag for an unswept wing mounted on an ogive-cylinder body for Mach numbers from 0.7 to 1.6 are presented in Technical Note 3550. The basic wing had an aspect ratio of 3.11, a taper ratio of 0.423, and 4-percent-thick circular-arc sections. The trailing-edge thicknesses investigated were 0, ⅓, ⅔, and ⅜ of the maximum thickness. It was found that wing drag increased with trailing-edge thickness, whereas the trailing-edge base pressures appeared independent of the thicknesses used. The calculated wing drag compared favorably with the experimental.

Results of tests, reported in Technical Note 3548, of a rectangular wing of aspect ratio 2.7 with 6-percent-thick diamond sections through the aforementioned range show that the blunt trailing edge gives the higher drag because of trailing-edge suction throughout the test range.

The effects of drooped leading edges on the flow over delta wings are described in Technical Note 3614. Vapor-screen, pressure-distribution, and ink-flow tests were conducted at the Langley Gas Dynamics Laboratory on a series of semispan delta wings. The delta wings had semiapex angles of 15°, 22.5°, and 31.75° and 10 and 20 percent of the semispans drooped 15° in streamwise sections. The tests were made at a Mach number of 1.9 and indicate that flow separation occurred on all the wings in the series tested. The separated regions on the wings with 10 and 20 percent of the semispans drooped were similar to one another. Integrated pressure distributions show that, for equal angles of attack, the loadings on the wings with 20 percent of the semispans drooped were less than those on the wings with 10 percent of the semispans drooped. In a general comparison with undrooped delta wings, the drooped-leading-edge wings show no particular advantage from a standpoint of preventing separation. The drooped-leading-edge wings had a disadvantage in loading, the loading in some cases being considerably less than that on the corresponding undrooped wings.

A knowledge of the effectsjof sweep and Mach number on wing aerodynamic characteristics near maximum lift is becoming more important as the speeds and altitudes flown by modern aircraft continue to increase. High-speed, high-altitude aircraft fly at rather high lift coefficients and in maneuvers may reach or exceed the angle of attack for maximum lift of the aircraft. Since sweptback wings are being used to delay and to minimize the effects of compressibility on some aircraft, it is important that the effects of sweep on the maximum lift coefficient are known. Considerable data are available for both swept and unswept wings up to maximum lift at low Mach numbers, but only a limited amount is available above a Mach number of approximately 0.60.

An investigation at transonic speeds was therefore made in the Langly high-speed 7- by 10- foot tunnel to determine the effect of wing sweep on the maximum-lift characteristics of a series of wings having aspect ratio of 4, taper ratio of 0.6, and NACA 65A006 airfoil sections. The Mach number varied from 0.61 to 1.20. It was found, as reported in Technical Note 3468, that the maximum lift coefficients increased with increased sweep at the lower Mach numbers but decreased with increased sweep at the higher Mach numbers. This resulted in less variation of the maximum lift coefficient with Mach number as the sweep was increased.

Several other studies concerned with the effects of wing plan form have been reported recently. For example, an investigation reported in Technical Note 3671 was conducted on the transonic bump of the Ames 16-foot high-speed wind tunnel to determine the effect of clipping the tips of a triangular wing. Four basic triangular-wing plan forms having aspect ratios of 2.0, 2.5, 3.0, and 4.0 were tested. The tips of these wings were progressively clipped to provide taper ratios of 0.1, 0.2, 0.3, 0.4, and, in some cases, 0.5. The NACA 63A004 profile was used with thickness-to-chord ratios of 0.02, 0.04, and 0.06. Lift, drag, and pitching-moment data were obtained for Mach numbers from 0.60 to 1.10, corresponding to test Reynolds numbers, from 1.85×10^6 to 2.90×10^6. For aspect ratios of 2.0 or greater the wings with pointed tips had lift-curve slopes that were consistently lower than those for the wings with clipped tips. Values of drag-rise factor were higher in each case for the wings with taper ratios of 0 than for any other values of taper ratio. Generally, the most significant decreases in drag-rise factor were realized when the taper ratio was increased from 0 to 0.1. The cambered wings had consistently lower values of drag-rise factor than the uncambered wings. In general, the values of the pitching-moment curve slope became less negative with increasing taper ratio. All wings tested exhibited a characteristic rearward shift of the center of pressure in going from subsonic to supersonic speeds. In general, decreasing the taper ratio decreased this rearward shift of the center of pressure.

In another study the effect of taper ratio on the zero-lift drag of a sweptback wing has been determined using rocket-powered models. Theoretical calculations were made for Mach numbers from 1.2 to 1.8 and were found to be in good agreement with the experimental data. The results are presented in Technical Note 3697.

Bodies

The Langley Pilotless Aircraft Research Division has conducted an investigation to determine the drag of practical fuselage shapes at transonic and supersonic speeds. One phase of this program is an investigation of how changes in nose shape affect the drag of an airplane or missile configuration. Linearized theory and some experimental data have indicated that, for minimum drag at supersonic speeds, the fuselage nose profile must be of high fineness ratio and tapered to almost a point at the vertex. It is of particular interest to determine how far practical designs can deviate from such profiles without severe reductions in speed and range.

In Technical Note 3549, drag data obtained during this study are presented for fin-stabilized bodies of revolution whose noses, originally pointed with fineness ratios about of 3½, have been rounded off with radii equal to ¼ the maximum body radii. By comparing the measured values of drag coefficient based on frontal area for the blunt- and pointed-nose models, it is found that, within the accuracy and range of the present tests, rounding off the sharp noses produced no increase in the total drag of either configuration.

Considerable interest exists at the present time in the drag characteristics at supersonic speeds of nonlifting bodies of revolution designed for minimum wave drag. One such family of boattail bodies has been investigated previously to assess the effect of Reynolds number and Mach number upon the wave-drag characteristics.

In order to find body profiles which have minimum total drag, the base-pressure drag (if no jet exists at the body base) and the skin-friction drag must also be considered. The results of experimental measurements made in the Langley 9-inch supersonic tunnel, presented in Technical Note 3708, show that the base drag and, in general, the total drag, increase with increasing values of the ratio of base area to the maximum cross-sectional area B/S_{max}. The results also show that the laminar skin-friction drag is in agreement with the theoretical predictions used, and, within the Mach number range of the tests (1.62, 1.93, and 2.41), the simple Blasius incompressible theory gives a satisfactory prediction. Except for the values of B/S_{max} near 1, the transition Reynolds number increases with increasing Mach number and, as this ratio approaches 1, the variation reverses. These variations in Reynolds number of transition with Mach number appear to be associated with changes in the pressure gradient over the rear of the bodies.

The shapes of nonlifting bodies of revolution having minimum pressure drag at supersonic speeds have also been the subject of numerous theoretical investigations. In Technical Note 3666, Newtonian impact theory is used in combination with the calculus of variations to determine body shapes for minimum pressure drag, neglecting base drag, at high supersonic airspeeds. Shapes are determined for various combinations of given length, base diameter, surface area, and volume. In addition, an estimate is made of centrifugal forces and their effects on the minimum-drag shapes were considered. In order to check the analysis, an experimental investigation was conducted in the Ames 10- by 14-inch supersonic wind tunnel on a family of bodies, including two of the minimum-drag shapes. The test results were found to substantiate the theoretical analysis.

A body moving at supersonic speeds has a wave drag which can be calculated either from integrations based upon the pressure at the surface of the body or by means of a momentum balance over a control surface surrounding the body. The control-surface approach shows more clearly that the wave drag is related to the transport of momentum in the Mach waves created by the body. This approach also suggests the scheme of reducing or destroying the wave drag through the use of a shroud as first shown by Ferrari. With such a shroud, the waves are caught and reflected to the body surfaces where they may be absorbed without further reflection. From the standpoint of the pressure exerted on the body itself, it follows that the reflected waves may strike the rear portion of the body in such a way as to provide a buoyancy to overcome the resistance of the body alone.

In Technical Note 3718, formulas for the wave drag of shrouded symmetrical airfoils and shrouded bodies of revolution of arbitrary shape are derived. The airfoil is shrouded by flat plates and the body of revolution is shrouded by a cylindrical shell. Although many configurations are possible, this analysis considers the particular arrangement where the shroud extends at least far enough forward to catch the Mach wave emanating from the body nose and far enough rearward to cast Mach waves on the base of the body. As a special application of the results obtained, a class of body shapes, similar to those given by Busemann and Ferri, was found for which the wave drag is theoretically zero.

Wing-Body Combinations

As part of a transonic-speed research program recommended by the NACA special subcommittee on research problems of transonic aircraft design, a systematic investigation of the effects of wing thickness and plan form on the aerodynamic characteristics in the transonic range has been conducted in the Langley highspeed 7- by 10-foot tunnel by the transonic-bump method.

A summary of the results of this investigation, presented in Technical Note 3469, indicates that, for subsonic Mach numbers below the force break, theoretical lift-curve-slope calculations were in fair agreement with the experimental results. In the supersonic range, however, the theoretical values were considerably higher than the experimental values. Increasing the thickness ratio caused rather large losses of lift in the transonic speed range, and increasing the sweep angle decreased these losses. Decreasing the thickness ratio and increasing the sweep angle increased the drag-rise Mach number and reduced the pressure drag. In general, the drag due to lift was increased by decreases in thickness ratio, increases in sweep angle, and decreases in aspect ratio.

The effect of sweep on delaying the onset of compressibility drag has generally been somewhat less beneficial than that indicated by simple sweep theory. This is caused, for the most part, by an adverse pressure distribution at the root of swept wings. A body-contouring method for alleviating this adverse interference at the root of a high-aspect-ratio swept-back wing has been studied in the Ames 14-foot wind tunnel. In this method the design objective is to alter the body shape so that the pressure distribution at zero lift, for a given subsonic design Mach number, will be the same at the wing-body junction as that on an infinite yawed wing at the same Mach number. This method of body contouring is primarily concerned with eliminating the interference at the root of a swept wing and should not be confused with the area rule which alters the axial distribution of cross-sectional area so as to minimize the wave drag at near sonic or supersonic speeds. Results of the study, presented in Technical Note 3672 show that beneficial results were obtained at free-stream Mach numbers above the critical, although modifying the body shape did not significantly affect the aerodynamic characteristics at subcritical speeds. Improved aerodynamic characteristics were evidenced by large reductions of drag, an increase in lift-curve slope, and a reduced change of pitching-moment-curve slope with increasing Mach number.

In Report 1252 a theoretical method developed at the Ames Laboratory is presented for calculating the supersonic flow field about wing-body combinations of bodies deviating only slightly in shape from a circular cylinder. If the combinations possess horizontal planes of symmetry, no restrictions are required on wing plan form for the zero-angle-of-attack condition; if the combination is lifting, the method requires that the flow over the wing leading edges be supersonic. The method was applied to the calculation of the pressure field acting between a circular cylindrical body and a rectangular wing. An experiment was performed especially for the purpose of checking the calculated examples. For very small angles of attack the method predicted the pressure distribution with good accuracy. For higher angles, nonlinear effects of viscosity and compressibility were encountered.

In problems involving interference drag, the detailed pressure distributions are usually not calculated. In Technical Note 3674, slender-body theory is used to calculate the pressure distribution for some nonlifting wing-body combinations in subsonic and supersonic flow. It was found that when the body is indented so that a constant total area of the cross sections normal to the stream is maintained, the pressures over the wing remain small throughout the transonic range and the isobars tend to remain smooth and nearly parallel to the sweep angle of the wing.

The flow fields and shock formations about a wing-body combination and its equivalent body of revolution as determined by the transonic area rule must correspond if there is to be a correspondence between the drags of the two bodies. Some limited information on the flow fields indicated that such a correspondence existed. In the absence of information whereby a comparison of the complete flow fields past a wing-body combination and its equivalent body could be made, tests utilizing the schlieren method of flow photography have been conducted in the Langley 4- by 19-inch tunnel. In Technical Note 3703, the flow fields past an unswept- and a swept-wing-body combination are compared with the flow fields past their equivalent bodies of revolution at Mach numbers around 1.0. The results of the tests indicate that the flow past the equivalent body of revolution duplicates, or closely approximates, that past the wing-body combinations. The results indicate, furthermore, that, as the bump representing the wing area on the equivalent body departs radically from a slender bump or small disturbance, the similarity of flow between the wing-body combination and its equivalent body decreases.

AERODYNAMIC STABILITY AND CONTROL

Static Stability

Considerable information is available on the influence of the wing, fuselage, and tail geometry on the static stability characteristics of high-aspect-ratio unswept-wing configurations. However, there is little information of a systematic nature on the effects of wing aspect ratio for complete models. An experimental investigation has therefore been made at low speed in the Langley stability tunnel to determine the effects of the various components and combinations of components on the static longitudinal and static lateral stability characteristics of models having unswept wings with aspect ratios of 2, 4, or 6. The results of this investigation, presented in Technical Note 3649, show that, at angles of attack near maximum lift, there was a pronounced increase in longitudinal stability for all wing-body combinations. At low and moderate angles of attack, decreasing the wing aspect ratio decreased the tail contribution to longitudinal stability. For the complete model, changes in wing aspect ratio generally had little effect on the tail contribution to directional stability throughout the angle-of-attack range investigated. The most noticeable effect of wing aspect ratio on the lateral stability characteristics occurred for configurations involving wing-fuselage combinations having an aspect ratio of 2. These configurations showed abrupt variations in the sideslip derivatives occurring at small angles of sideslip for angles of attack of about 20° and 26°, respectively.

A large decrease existed in the vertical-horizontal tail contribution to directional stability at moderate and high angles of attack. This was caused by the effect of wing-fuselage interference at the tail which was counteracted by a stable shift in the directional stability of all wing-fuselage combinations investigated. As a result of these combined effects, all complete-model configurations were directionally stable throughout the angle-of-attack range investigated.

The stability derivatives of such midwing research models which have simple bodies of revolution can, in general, be estimated with good accuracy in the low angle-of-attack range by various theoretical and empirical methods. Unpredictable interference effects caused by the addition of ducts, canopies, or other protuberances, makes the estimation of the stability derivatives more difficult and often impossible.

These models were modified to find the effects of size and position of closed wing-root air ducts on the static longitudinal and static lateral stability characteristics. In addition, the effects of top and bottom fuselage ducts on these characteristics were found for model configurations employing the wing having an aspect ratio of 2.

These results, reported in Technical Note 3481, show that, in the low angle-of-attack range, the addition of and increase in size of the wing-root ducts on model configurations employing the unswept wing with an aspect ratio of 2 resulted in a large forward movement of the aerodynamic center. This forward movement resulted regardless of whether the horizontal tail was located at the base or at the tip of the vertical tail. Increasing the wing aspect ratio from 2 to 6 made this effect more pronounced. A slight rearward movement in the areodynamic center resulted from the addition of and increase in size of the top and bottom fuselage ducts. Regardless of the aspect ratio of the wing, the addition of and increase in size of the wing-root ducts caused an increase in directional stability for the complete models at low angles of attack. The addition of and increase in size of top and bottom fuselage ducts on the complete model with the aspect-ratio-2 wing, however, resulted in a large decrease in directional stability which was about constant throughout the angle-of-attack range investigated.

In another study the effect of fuselage cross-sectional shape was determined for models having 0° or 45° sweptback wing. The results, presented in Technical Note 3551, show that the direct contribution of the fuselage cross section affects the longitudinal and directional stability characteristics at low angles of attack. At high angles of attack, in addition to the direct contributions of the fuselage, wing-fuselage interference (sidewash) with the tail decreases the directional stability. The configuration with the shallow fuselage suffered the least from this detrimental effect. In general, the configuration with a deep fuselage had the poorest directional characteristics of the models investigated.

One longitudinal stability problem of particular concern with respect to swept-wing airplanes is the "pitch up" which manifests itself essentially in a reversal of the variation of elevator control position and force with normal acceleration. This pitch-up behavior, as far as the pilot is concerned, limits the useful maneuvering range of the aircraft. Tests are continuing, therefore, so that this condition can be better understood and eliminated. In one Ames Laboratory study, flight tests were conducted on a 35° swept-wing aircraft to determine the origin of the pitch up. The results, presented in Report 1237, show that the pitch up encountered in a turn at constant Mach number was caused by an unstable break in the wing pitching moment with a corresponding increase in lift. The pitch up encountered at about 0.95 Mach number in a dive-recovery maneuver was due chiefly to a reduction in the wing-fuselage stability with decreasing Mach number. The unstable break in the wing pitching moment has been linked in part to separated flow over the wings. This can also result in buffeting and wing dropping.

Among the modifications studied at the Ames Laboratory to reduce the effects of flow separation, vortex generators, a development by H. D. Taylor of the United Aircraft Corporation, are shown in Technical Note 3523 to be effective. These devices are small airfoils placed perpendicular to the surface in a flow field in such a manner as to create vortices with axes alined in the flow direction. Properly designed vortex generators thus provide intermixing of the retarded boundary-layer flow with the higher energy flow further from the surface and, hence, tend to delay separation. Vortex generators of proper design were effective in delaying the lift coefficient at which the pitch up occurs as well as in reducing the severity of wing dropping.

Considerable research effort has also been devoted to improving undesirable pitching-moment characteristics of swept wings by use of slats, fences, and other devices. In one flight investigation made at the Ames Laboratory with a 35° swept-wing airplane, the effects of a partial-span, 15-percent-chord leading-edge extension were evaluated. The extension was found to be highly effective in eliminating the stick-fixed instability (pitch up) for Mach numbers below 0.84, though no benefit was observed at Mach numbers between 0.84 and 0.88. For Mach numbers above 0.88, the lift coefficient at which the pitch up commenced was increased somewhat but the severity was not significantly changed. Pilot opinion indicated improvement in the modified airplane flight characteristics at Mach numbers at which the pitch up was eliminated.

In another flight investigation, undertaken at the Ames Laboratory, the effect on the outboard wing sections of reducing the trailing-edge angle by using blunt trailing-edge ailerons was determined. Wind-tunnel tests had indicated that reduced trailing-edge angles would be beneficial. The results show significant improvement in the pitch-up characteristics at Mach numbers around 0.90.

Another strong influence on the longitudinal stability of an airplane is the downwash. Until recently, little information of a systematic nature was available relative to the effects of wing geometry on the downwash characteristics at transonic speeds. As part of a transonic research program, however, the effects of changes in wing plan form and thickness were investigated through a Mach number range of approximately 0.6 to 1.1 by utilizing the Langley high-speed 7- by 10-foot tunnel transonic bump. Downwash information was obtained by the use of floating vanes mounted at various positions behind eleven wings of various plan forms and profiles. Theoretical estimates of the downwash were made for all of the wings at subsonic and low supersonic speeds by use of compressible vortex theory and 20-step wing spanwise loadings. Reasonable agreement between theory and experiment was obtained except for the chord-plane location at subsonic speeds. It was found that the agreement for this condition could be improved by assuming a completely rolled-up vortex sheet. Results are presented in Technical Note 3628.

The effects of wing sidewash on the vertical tail can also be of considerable importance. In Technical Note 3609, results are presented of a theoretical analysis, conducted in the theoretical aerodynamics section of the Langley Stability Research Division to determine by means of linearized lifting-surface and lifting-line theories the sidewash field behind rolling triangular wings with subsonic leading edges for supersonic flight speeds. The analysis also includes the development of an expression for the sidewash based on horseshoe-vortex approximate-lifting-line theory. The approximate procedure, which is useful in computing the sidewash for wings of arbitrary plan form and span loading is in good agreement, in general, when compared with the more exact calculations for the triangular wing. Variations of the sidewash with longitudinal distance in the vertical plane of symmetry are presented in graphical form; illustrative calculations of the resultant induced force acting on various vertical tails are also included.

It has been recognized for some time that interference among the various airplane components can be very important in determining the aerodynamic characteristics of slender configurations. The use of slender-body theory for treating entire wing-body combinations is not new, and for cases falling into this category considerable simplification has resulted. The calculation of interference effects due to wing wakes (wing-tail or wing-afterbody interference), on the other hand, is not so clear cut. Unlike the wing-body problem, this calculation becomes more difficult as the effects become more important, since the rolling up and displacement of the wing vortex sheets cannot be ignored for long slender configurations.

In Technical Note 3525, formulas are developed for the forces and moments caused by vortex interference on slender wing-body-tail combinations of general cross section in terms of the positions and strengths of the shed vortices. The analysis is applicable to steady motion and to motions which can be considered as a succession of steady states (i. e., quasi-steady motions). In order to illustrate the application of the analysis, the interference lift of a plane wing-body-tail combination in steady straight flight was determined by utilizing vortex positions obtained by numerical methods. It was found that the impulse of each shed vortex and its image vortex in a transformed circle plane enter into all the interference forces and moments on the airplane. A simple theorem is given for the interference forces in steady straight flight; these forces are found to depend on this impulse evaluated only at the wing trailing edge and at the base of the configuration.

In another theoretical study, the pressures, loadings, forces, and vortex wake associated with low-aspect-ratio cruciform wing arrangements were determined. For 45° bank, the wake of a cruciform wing is treated numerically with 40 vortices and analytically with 4 vortices. Comparisons are made with water-tank measurements. Formulas are also given for the calculation of loads on a cruciform tail operating in the wake of a wing. These results are reported in Technical Note 3528.

A method (described in Technical Note 3670) has also been developed by the Ames Laboratory for calculating vortex paths. The method is applicable to any

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situation where the initial positions and strengths of the vortices are known. The method was then applied to computation of vortex positions behind a slender equal-span cruciform wing at several different bank angles and from these computed positions the interference forces on a tail were calculated. Comparison with the known analytical solution for 45° bank angle shows good agreement.

In exploring the possibilities for the attainment of long range in airplanes designed to fly at high subsonic speeds, consideration has been given to the aerodynamic problems associated with the use of turbine-driven propellers operating ahead of a sweptback wing of high aspect ratio. The effects of operating propellers on the stability and control of multiengined aircraft having unswept wings have been quite thoroughly investigated in past years. The newer configurations, however, involve several elements which make the results of the earlier work either inapplicable or inadequate. Of paramount importance is the large increase in propeller disk loading (with the associated increase in slipstream intensity) which, in combination with wing sweepback, might be expected to create longitudinal stability difficulties.

A related investigation was carried out in the Ames 12-foot pressure wind tunnel to evaluate the effects of operating propellers on the longitudinal aerodynamic characteristics of a representative four-engine airplane configuration with a 40° sweptback wing. The wind-tunnel model was designed to simulate an airplane capable of long-range operation at a cruising speed of 550 miles per hour at an altitude of 40,000 feet with very high wing loadings. The thrust of the single-rotation propellers was sufficient to simulate up to 10,000 horsepower per engine at sea level, assuming the model to be 1/12 scale. The tests were conducted at Mach numbers from 0.08 to 0.92 and included measurements of lift, longitudinal force, pitching moment, propeller thrust, and propeller power for a number of model configurations. These results, published in Technical Note 3789, show that for the configuration investigated, the use of propellers presents no serious longitudinal-stability problems at high speeds nor is the drag or the Mach number for drag divergence adversely affected by propeller operation. At low-speed conditions corresponding to take-off and landing, however, destabilizing effects of operating propellers may be very large, requiring, therefore, careful attention in aerodynamic design. The measurements were in sufficient detail to permit determination of the origin, nature, and intensity of the various components of the total measured effects of the operating propellers. Such an analysis has been published in Technical Note 3790, along with recommendations for reducing or avoiding adverse effects of propellers on similar configurations.

With the trend of modern high-speed aircraft, particularly missiles, toward the use of tail configurations incorporating surfaces of low aspect ratio, a number of theoretical papers have been published on the determination of the stability characteristics of this type of configuration. Although most of these papers are concerned primarily with supersonic flow, some, which are based on slender-body theory (i. e., covering thin wings of extremely low aspect ratio and slender bodies), are applicable also at subsonic speeds. The theoretical stability characteristics for some tail configurations have been verified experimentally; however, only meager or no experimental data exist for a range of V-, Y-, and cruciform-tail configurations, particularly at low speeds. A low-speed experimental investigation was therefore made in the Langley stability tunnel to determine the static lateral and static rolling stability derivatives of a series of cruciform, inverted T-, V-, and Y-configurations composed of low-aspect-ratio triangular surfaces. The derivatives are presented as functions of the geometric characteristics of the models in Technical Note 3532, and for two configurations (a planar wing and an inverted T), as functions of angle of attack. Where possible, comparisons have been made between experiment and existing theory; in general, it was found that better agreement between theory and experiment was obtained for the sideslip derivatives than for the rolling derivatives.

Dynamic Stability

As part of a general research program concerned with the lateral dynamic stability and handling characteristics of high-speed, high-altitude airplanes, the Ames Aeronautical Laboratory has tested a 35° swept-wing fighter airplane through a wide range of flight speeds and altitudes. In Technical Note 3521 are presented results of tests of the lateral oscillatory characteristics made during a series of four flights. Comparisons are included of the computed variation of period and damping of the lateral oscillation with the measured values. These comparisons indicate the accuracy with which the oscillatory behavior of an airplane can be predicted under various flight conditions using available or estimated mass parameters and stability derivatives and neglecting such effects as aeroelasticity and unsteady lift.

The airplane was found to be laterally stable, statically and dynamically, throughout the speed range tested. The variation with Mach number of the period of the lateral oscillation was satisfactorily predicted from available and estimated aerodynamic and mass parameters. The time required to damp to half amplitude, as measured from flight, varied with Mach number in essentially the same manner as predicted from computations. The measured damping was somewhat better than that obtained from computations at an altitude of 35,000 feet, particularly at a Mach number of 0.92. An increase in time to damp to half amplitude was noted between Mach numbers of 0.95 and 1.04.

Several modern high-speed airplanes incorporate structural components that are more flexible than those previously used. There has been concern that the increased flexibility might appreciably modify the dynamic stability characteristics as predicted by rigid-airplane theory. Particular concern was felt for the possibility of interaction between structural and stability vibratory modes because the natural frequencies of the major structural components are approaching the natural frequencies of the short-period stability mode.

The problem of the effect of wing flexibility on the dynamic longitudinal stability of large airplanes has already been treated analytically in the subcritical speed range by a simplified semirigid method. In Technical Note 3543, the effects of fuselage flexibility are studied by the same method for the same class of airplanes.

Results of the study indicate that no serious problems are introduced insofar as the dynamic longitudinal stability of high-speed bomber airplanes in the subcritical speed range is concerned. The results further indicate that, if desired, future designs may incorporate somewhat more flexible fuselages than the typical current design studied and still have dynamic longitudinal characteristics approximately equal to those predicted by quasi-static theory and roughly equivalent to those predicted by rigid-body theory. The aerodynamic restoring forces which act on the airplane are sufficient to increase the frequency of the fuselage oscillatory mode in flight so that the latter frequency is always higher than either the fuselage-ground structural frequency or the airplane-stability natural frequency. Thus, the occurrence of a resonant condition is avoided. The changes in static characteristics caused by increased fuselage flexibility appear as decreased straight-flight and increased maneuvering-flight stability margins and increased elevator control deflections required for balance. The need for a means of longitudinal balance in steady maneuvering flight, other than elevator or horizontal-tail deflection, is indicated if flight is desired at low altitude and high speed with large airplanes that have fuselage natural frequencies much below those representative of current design practice.

In the design of automatic-control equipment for high-performance aircraft, the dynamic response characteristics of the aircraft must be considered. Often these dynamic characteristics can be predicted by using stability derivatives obtained from wind-tunnel tests. In many cases, however, particularly in the transonic speed range, flight-test procedures are desirable to document the dynamic behavior of the airplane. Flight tests also serve the additional purpose of enabling comparisons to be made with predicted results, thus aiding in the development of more refined prediction methods.

A related flight investigation has been conducted by the Ames Laboratory in which the longitudinal and lateral-directional dynamic-response characteristics of a 35° swept-wing fighter-type airplane were evaluated from Mach numbers of 0.50 to 1.04. The results of the study are given in Report 1250. Responses to transient rather than sinusoidal control inputs were chosen for analysis because of convenience in making flight measurements. These transient data were converted into frequency-response form by means of the Fourier integral and compared with predicted responses calculated from the basic equations of motion. The equations, or transfer functions, that best describe the various measured responses were evaluated by a curve-fitting process involving the use of templates and an analog computer. By this method it was generally possible to find equations, of simple form, that closely matched the experimental frequency responses between 1 and 10 radians per second and at the same time adequately described the recorded time histories. Experimentally determined transfer functions were used for the evaluation of the stability derivatives that have the greatest effect on the dynamic response of the airplane. The values of these derivatives, in most cases. agreed favorably with predictions over the Mach number range of the tests.

Recent work on wing configurations designed to redirect propeller slipstreams downward has demonstrated that this principle can be used to provide direct lift for vertical take-off and landing. With transport aircraft, using this principle, it is possible to keep the fuselage approximately horizontal at all times.

The program of the Langley 7- by 10-foot tunnels which is concerned with the study of the promising double-slotted-flap configurations, has been extended to determine the effect of ground proximity and to develop a leading-edge longitudinal-control device. In Technical Note 3629 it is shown that, with the propeller-thrust axis on the wing-chord plane, both the slipstream deflection and the ratio of resultant force to thrust are reduced as the ground is approached. It is also shown that lowering the thrust axis below the wing-chord plane tends to alleviate these adverse ground effects while at the same time reducing the large diving moments associated with the slotted-flap configuration. Technical Note 3692 presents results which indicate that, for configurations utilizing deflected slipstreams, a leading-edge slat, preferably above the wing-chord plane, can provide increments of pitching moment of the order of those required for control and change in trim with center-of-gravity travel for a vertically rising airplane in hovering flight. In the ground-effect region, however, the slat is generally ineffective as a longitudinal-control device.

In another Langley investigation, the stability and control characteristics of a low-wing four-engine transport vertical-take-off airplane have been determined with a remotely controlled free-flight model. In order to provide direct lift for hovering flight with the fuselage horizontal, the wing and propellers were rotated 90° with respect to the fuselage. Despite the fact that the pitching and rolling motions of the model were unstable oscillations, the model could be flown smoothly and easily without the use of any automatic stabilization devices because the periods of the oscillations were fairly long and the controls were powerful. The pitching oscillation could be completely stabilized by the use of artificial damping in pitch; thus the model could be flown in pitch for long periods of time without the use of the manual pitch control. Although there was no stability of yaw position, the model was easy to control in yaw because the motions were slow and the yaw control was powerful. There were no noticeable interactions between the rolling and yawing motions or between the roll and yaw controls. Vertical take-offs and landings could be performed fairly easily, although some forward or backward motion of the model was often present. Results of the study are given in Technical Note 3630.

Damping Derivatives

One of the more important factors entering in aircraft stability and control calculations is the aerodynamic resistance to roll or damping in roll. The damping in roll is generally expressed in terms of the nondimensional parameter C_{l_p} which is the rate of change of rolling-moment coefficient with change of wing-tip helix angle, $pb/2V$. To check the validity of theoretical results, data were obtained by means of a forced-roll technique which consisted of rolling various wings in the Langley 9-inch supersonic tunnel at various constant rolling velocities and then measuring the aerodynamic resistance. Some of the results for the rectangular and triangular plan forms, obtained at Mach numbers of 1.62 and 1.92, are reported in Technical Notes 3740 and 3745. Comparisons of the experimental results with predictions based on linearized theory indicate that the damping in roll was predicted accurately for rectangular wings and for all other wings at Mach numbers for which the leading edge was decidedly supersonic (leading edge swept well ahead of the Mach trace emanating from the apex). When the leading edge was substantially subsonic (leading edge swept well behind the Mach trace), theory satisfactorily predicted the damping for triangular wings but appeared to overestimate (in some cases considerably) the values obtained for the sweptback, tapered wings. In the transition range, where the leading edge was near sonic, the theory, as expected, overestimated the damping in roll for all applicable plan forms.

As part of a continuing investigation of the effects of unsteady motion on the lateral stability derivatives of airplane models, tests were made in the Langley stability tunnel at low speeds to determine the effects of frequency and amplitude on the yawing moment due to rolling and the damping in roll for an unswept-wing airplane model. These tests, which were preliminary in nature, involved the forced oscillation in roll of the model about its longitudinal wind axis through a range of frequencies and amplitudes of motion. The results obtained were primarily for an angle of attack of 0°. Steady-state derivatives were measured by means of tests made with the model stationary in rolling flow and with the model rolling steadily at several rotary velocities in straight flow. These steady results are regarded as zero-frequency oscillation data and form the basis for a comparison of the unsteady-state and the steady-state rolling derivatives. Theoretical values for the steady-state rolling derivatives were also used for comparison with the experimental data.

The results of the investigation, presented in Technical Note 3554, show that, in the range covered, variations in the frequency and the amplitude of oscillation at 0° angle of attack had no important effect on either the yawing moment due to rolling or the damping in roll of the complete model. The fuselage-tail combination experienced a reduction in the yawing moment due to rolling as either the frequency or amplitude was increased. The values of the rolling derivatives obtained by oscillation methods were consistent with the values measured by means of rolling-flow tests at an angle of attack of 0°. At a high angle of attack, the model with the wing had a different oscillatory yawing moment due to rolling when compared with that determined under steady-state conditions.

Control

Some effects of wing geometric characteristics, aeroelasticity, and Mach number (from 0.6 to 1.8) on the rolling effectiveness of various types of ailerons have been obtained from a general investigation of lateral controls by the Langley Pilotless Aircraft Research Division with the use of free-flight rocket-propelled models. For plain, sealed, 0.2-chord, full-span, flap-type ailerons, the control effectiveness at supersonic speeds was found to be markedly less than at subsonic speeds for all the configurations tested; however, increasing the wing sweepback decreased the abruptness of the change in rolling effectiveness between subsonic and supersonic speeds and reduced the general level of rolling effectiveness throughout the speed range. Furthermore, the effectiveness increases with decreasing aspect ratio. Tapering the wing was found to have a beneficial effect on control effectiveness for unswept wings but somewhat detrimental effect in the transonic region for wings swept back 45°. It was also shown that increased control effectiveness in the transonic region resulted from reducing the section thickness from 0.09 to 0.06 for an unswept wing but had little or no effect upon a 45° swept wing.

It was also found that small changes in the shape of the forward part of the airfoil do not appreciably change the control effectiveness, but trailing-edge angle has a powerful effect upon the control effectiveness; that is, positive control was obtained over the Mach number range when the airfoil-section trailing-edge angle was of the order of 10° or less; control reversal was encountered when the trailing-edge angle was of the order of 16° to 20°. A convenient way to achieve the beneficial effects of a small trailing-edge angle is to use slab-sided ailerons with thickened trailing edges. Camber was found to have little or no effect upon the rolling effectiveness of an outboard partial-span aileron on an inverse tapered swept wing.

Results of tests of a thin unswept wing with outboard partial-span ailerons show a very rapid decrease in control effectiveness with increasing Mach number above about 0.9. At supersonic speeds, the rolling effectiveness was only a small fraction of that obtained for a Mach number of 0.9. Tests with steel and aluminum-alloy wings indicate that the losses in control effectiveness due to aeroelasticity were negligible for this configuration. Other tests of a swept-tapered-wing research airplane with outboard partial-span ailerons show severe losses in control effectiveness (including control reversal) because of wing flexibility when scale wing stiffness was considered. Increasing the sweepback angle of the wings increased the Mach number for control reversal.

Results of tests of delta wings with ailerons show that the rolling effectiveness of constant-chord full-span flap-type ailerons on 45° and 60° swept delta wings and a delta-wing tip aileron on a 45° swept wing were about the same at Mach numbers less than 0.9. All controls exhibited a loss in effectiveness at supersonic speeds, but the rolling effectiveness of the delta-wing tip ailerons was about twice that of the constant-chord ailerons.

The control forces on aircraft operating at supersonic speeds are so high that very substantial power-boost systems are usually required to handle the hinge moments. In an attempt to find a solution to the problem of reducing the size and work requirements of boost systems for such aircraft, theoretical analyses have been made of the hinge moments due to deflection of unbalanced trailing-edge flap-type controls. These controls had plan forms varying throughout the range in which the control leading and trailing edges are supersonic and the control tips are streamwise. Ratios of lift and rolling moment to hinge moment and ratios of lift and rolling moment to deflection work at fixed values of lift and rolling effectiveness were used as bases for the analyses, reported in Technical Note 3471.

Results of the analyses for longitudinal controls show that high-aspect-ratio untapered controls possess maximum ratios of lift to hinge moment. When low-aspect-ratio controls must be used, however, controls with triangular plan forms and highly swept hinge lines are shown to have higher values of the ratio of lift to hinge moment than untapered controls. Ratios of lift to deflection work for untapered controls are in most cases shown to be higher than those for controls with tapered plan forms.

On wings with sweptforward and unswept trailing edges, inversely tapered controls with triangular plan forms of moderate or low aspect ratio are shown to have maximum ratios of rolling moment to hinge moment. On wings with sweptback trailing edges, maximum values of this ratio are shown for either untapered or normally tapered controls. For any given control shape, the analysis illustrates the importance of using small controls with high deflections to obtain large values of rolling moment to hinge moment ratio.

Maximum ratios of rolling moment to deflection work on wings with sweptforward trailing edges are in most cases obtained with inversely tapered controls with triangular plan forms. On wings with unswept and sweptback trailing edges, the deflection work required is near minimum for untapered controls with spans of about two-thirds the wing semispan. Results indicate that large controls will in most cases have higher ratios of rolling moment to deflection work than smaller controls.

Another approach to this problem is to link the trailing-edge control to a leading-edge flap to cancel hinge moments. A theoretical study of such linked controls operating at supersonic speeds has been made in the Langley 19-foot pressure tunnel. The results reported in Technical Note 3617 indicate that substantial reductions in hinge moment can be realized, particularly for sweptback wings.

Another related investigation was made in the Langley high-speed 7- by 10-feet tunnel to determine the feasibility of using a servovane control at high subsonic and low transonic speeds. This servovane control, located ahead of and geared to a flap-type control, could presumedly be deflected with relatively low control forces compared with flap-type controls. The control was tested on an untapered semispan wing of aspect ratio 3 with 35° of sweepback. Throughout the speed range tested (Mach numbers from 0.6 to 1.0), increments of lift, pitching moment, and rolling moment were produced in the correct direction over most of the angle-of-attack range tested. At an angle of attack of 0° the servovane control gave a higher incremental lift coefficient and had a more forward location of center of pressure of lift than a comparable flap-type control. Results of the study are given in Technical Note 3689.

The emphasis on simplifying or eliminating power-boost systems required to move the controls of high-speed aircraft has led to consideration of using some part or all of the jet-engine air to provide control. In order to keep the quantity of air used to a minimum, a control system has been devised that obtains its effectiveness both from the reaction of the jet of air being ejected out of the wing and from the change in circulation about the wing resulting from the jet acting as a spoiler. The fact that a jet of air provides changes in lift similar to a plain spoiler has been known for some time, but the results have been limited to two-dimensional, very thick, airfoils. One advantage of this type of control is that an emergency control can be obtained by using air at stagnation pressure if the jet engine fails.

A related low-speed wind-tunnel investigation was made in the Langley 300 mph 7- by 10-foot tunnel of such a jet control as an aileron, a 35° sweptback wing having an aspect ratio of 4.76. The investigation was of an exploratory nature and was limited to the case where the jet was supplied with air at stagnation pressure. The results indicate that the air at stagnation pressure will provide adequate control for emergency flight for a system in which normal control is obtained by using a jet of air at high pressure or in which a spoiler is used in conjunction with the jet at stagnation pressure.

The spoiler used as a lateral-control device has been the subject of considerable investigation at low and high speeds, and on both swept and unswept wings. Recent investigations of spoilers used as lateral-control devices have shown that on thin wings with small leading-edge radii the unvented spoiler loses effectiveness rapidly as the angle of attack is increased above about 8°. It has been found that this loss in effectiveness at the higher angles of attack can be reduced at low speeds by using a slot in the wing behind the spoiler that allows the air to flow through the wing from the lower to the upper surface when the spoiler is deflected.

To determine whether a slot plus a deflector is as effective at high subsonic speeds as it was at low speeds, an investigation was conducted in the Langley high-speed 7- by 10-foot tunnel for a Mach number range from 0.40 to 0.91 of a 50-percent-semispan, inboard, spoiler-slot-deflector configuration on an aspect-ratio-4 wing with the quarter-chord line swept back 32.6°. The spoiler-slot deflector was located between the 55- and 70-percent-chord line. The results of the investigation indicate that the loss in rolling-moment effectiveness of an unvented spoiler at high wing angles of attack is materially reduced by the incorporation of a slot and deflector at Mach numbers up to 0.91. The optimum ratio of deflector to spoiler projection for best results varied with angle of attack, but a ratio of three-fourths to one gave appreciable rolling-moment effectiveness through the angle-of-attack range from 0° to 20°.

High Lift and Stall

Final results of an extensive investigation into the possibilities of obtaining reliable lateral control at the lowest flight speeds of light airplanes are reported in Technical Notes 3676 and 3677. Previous results were reported in Technical Note 2948. This investigation was conducted at the Texas Agricultural and Mechanical College under the sponsorship of the NACA. It was found that, for all of the aircraft tested, adequate lateral control is available up to some critical angle of attack that is always within 2° of the angle of attack for maximum lift. The elevator deflection required to trim at this condition has been found, with power off and power on, for each of the aircraft tested, as well as the elevator deflection required to make a three-point landing. Flight tests were also made with one airplane having two different horizontal tail configurations in an attempt to provide an arrangement that would give near-optimum conditions with regard to the effect of power change on longitudinal trim near the stall. This attempt was successful with one of the configurations tested, so that under all of the conditions of power setting and center-of-gravity position tested the available elevator deflection was sufficient only to maintain the angle of attack at a point where lateral control remained adequate. The increase in minimum speed was negligible.

Analytical means are provided by which designers may estimate the elevator deflection required to trim in steady longitudinal flight; the effects on longitudinal trim of changes in some of the more important design parameters are demonstrated in a quantitative manner. It is concluded that the procedures suggested can result in a design in which the maximum up-elevator deflection may be maintained within the highest value that will result in satisfactory damping in roll and reliable lateral control under all flight conditions, while, at the same time, adequate longitudinal control is available.

In view of the successful application of vortex generators for reducing or eliminating flow separation under certain conditions, a limited flight investigation was made to determine whether vortex generators might reduce the separation on a trailing-edge flap sufficiently to improve the flap effectiveness. The results, presented in Technical Note 3536, show that none of the three vortex-generator configurations tested provided any increase in flap effectiveness for flap deflections of 45°. With a flap deflection of 20°, one of the configurations gave an increase in lift at a given angle of attack equivalent to about an 8° increase in flap deflection. However, the drag was also increased by about the amount that would be caused by the equivalent increase in flap deflection. The vortex-generator arrangements tested, therefore, provided no net improvement in flap effectiveness from the standpoint of high-lift capabilities, although they appeared to offer some possibilities, through their effect at moderate flap angles, of improvement of flap-type control characteristics.

Spinning

In order to obtain a broader understanding of the factors which affect spin and recovery motions, a technique has been devised for determining time histories of the attitudes and velocities of free-spinning-tunnel models from film taken by two motion-picture cameras operating simultaneously. The method devised and the results of an initial application of the method in which time histories of attitudes and velocities have been determined for one medium-attitude, moderately oscillatory, developed spin and the recovery therefrom for a model representative of a contemporary fighter airplane, are described in Technical Note 3611.

The time-history curves indicate that the oscillatory motion of the spin was not completely regular, inasmuch as the period and amplitude of any one cycle of the time histories were not exactly the same as those for the preceding and following cycles. After rudder reversal from "with" to "against" the spin, some increases occurred in the amplitude of the oscillations, particularly in the lateral attitude angles of sideslip and wing tilt. The recovery was completed about 1½ turns after the rudder was reversed. One factor which appeared to have an important part in the recovery from the spin in this investigation was a relatively high inward sideslip reached during the oscillations following rudder reversal. Analysis indicates that the high inward sideslip apparently caused a relatively large negative aerodynamic rolling moment which caused a large decrease in rolling velocity. As a result, there was a corresponding large decrease in the gyrodynamic pitching moment which had been holding the model in a nose-up spinning attitude.

The results of an investigation made in the Langley 20-foot free-spinning tunnel to study the gyroscopic effects of jet-engine rotating parts (or of rotating propellers) on erect spin and spin-recovery characteristics are presented in Technical Note 3480. The angular momentum of the rotating parts was simulated on the model by a rotating flywheel powered by a model airplane engine. The rotating flywheel (rotating clockwise as viewed from the cockpit) generally caused the model to spin at a decreased angle of attack and an increased rate of rotation in right spins, and at an increased angle of attack and a decreased rate of rotation in left spins. The recovery characteristics generally changed from satisfactory to unsatisfactory for right spins. For left spins, however, the satisfactory recovery characterists obtainable with the flywheel not rotating were not appreciably altered when the flywheel was rotating. Results indicate that the effect of rotating the flywheel could be influenced by loading.

Research Equipment and Techniques

A simplified method for obtaining free-flight measurements of longitudinal aerodynamic characteristics of airplanes and missiles has been developed by the Langley Pilotless Aircraft Research Division for use with rocket-powered models. The basic principle of this method is that the horizontal tail of a configuration is mass balanced and hinged rearward of its aerodynamic center as an all-movable tail. A continuous pitching oscillation of approximately constant amplitude is sustained throughout the Mach number range as the tail automatically flips between stop settings when the tail lift changes direction. This technique has been applied experimentally to a rocket-powered model of an arrow-wing airplance configuration having a T-tail arrangement. Data were obtained for this configuration on the drag due to lift, lift-curve slope, tail effectiveness, and effective downwash at the tail. Both preflight calculations and flight-test data showed that downwash from the wing increases the angle of attack at which the tail will flip. The steady-state angle-of-attack response to a unit tail deflection, therefore, should be slightly greater than the required angle of attack to flip the tail in order to insure that a continuous pitching oscillation will develop.

Free-flying dynamic scale models, frequently used for securing aerodynamic research data, generally use movable aerodynamic surfaces in order to furnish disturbing or restoring forces and moments for purposes of investigating the response of the model. Often, these aerodynamic surfaces are in unsteady or unknown flow fields; consequently, the magnitudes of forces or moments being applied are not accurately known. The use of small rockets to produce the required disturbing forces or moments minimizes these difficulties. Rocket motors have been used with good results to disturb the flight of rocket-powered research models in pitch and yaw. Recently, a requirement for the disturbance of a free-spinning model in the Langley 20-foot free-spinning tunnel resulted in the development of a miniature rocket producing 3 ounces of thrust for 2 seconds. The engineering methods necessary to produce the desired characteristics and the steps of research and design necessary to fabricate such a rocket are presented in Technical Note 3620.

AUTOMATIC STABILIZATION AND CONTROL

The problem of determining the differential equations governing the behavior of a dynamical system, given a time history of the response of the system to some input, has been of interest to aerodynamicists for some years. A general theory for the analysis of dynamical systems has been developed at the Ames Laboratory ("equations-of-motion" method). This theory was presented at the Conference on Nonlinear Control Systems conducted jointly by the American Society of Mechanical Engineers and the American Institute of Electrical Engineers. It was noted that, when looked at from a new point of view, all such methods can be generalized so as to apply to linear and nonlinear systems alike. Use of this theory also has shown how new methods can be developed to satisfy the requirements of particular problems. One method developed has certain advantages over methods heretofore used. Its superiority is based on two facts: (1) The heavy dependence on the initial conditions which occur when using most of the previously known equations-of-motion methods is eliminated, and (2) there is no need for infinitely long records of the motions of the system following a disturbance. Finally, the time of application of the new method is no greater than that for existing methods and it is well suited to machine computation.

One of the factors that affects the precision of such flight operations as landing and air gunnery is the response of the airplane to gust disturbances. Therefore, it is highly desirable to study the motions of airplanes and airplane-autopilot combinations which result from gusts and to establish which combinations are best suited to minimize the motions. A related theoretical study has been made at the Langley Laboratory to determine the lateral response of a fighter airplane to atmospheric turbulence as approximated by side gusts and rolling gusts. The frequency response and power spectral density of the motions of the airplane with controls fixed and in combination with three different basic types of attitude autopilots are presented in Technical Note 3603. It was found that the airplane alone exhibited a large resonance to gust inputs associated with the Dutch roll mode. The addition of a yaw damper eliminated the resonance. The autopilot designed to provide regulation in yaw and roll greatly reduced the response to gusts. The autopilots designed to provide good course response to automatic lateral-steering command signals resulted in a large undesirable roll and yaw response to side gusts.

It was felt that this difficulty might not be basic and that the gust-response characteristics might be improved without deteriorating the command characteristics. A simplified theoretical study of methods to correct the undesirable gust response was therefore undertaken. The results of this study are presented in Technical Note 3635 which describes a simple modification to the yaw channel of an autopilot which reduces the roll response to side gusts while causing no change in course response. The study indicates that response to side gusts will be greatly reduced.

A theoretical study using an electronic analog computer has also been made of the response to step gusts of an airplane equipped with a system which operates to reduce accelerations in rough air. In this system, modified wing flaps and the elevator are actuated by an automatic control system in response to the indications of an angle-of-attack vane. The effect of interconnection of the flap-operating mechanism with the pilot's control system is also included. The analysis, given in Technical Note 3597, shows that modified trailing-edge wing flaps and a coupled elevator operated by a vane-controlled servo system are effective in reducing normal acceleration and pitching velocity due to step gusts when the flap-elevator system is adjusted to maintain a steady-state constant lift and zero pitching moment about the wing aerodynamic center.

It was also found that desirable dynamic stability and control characteristics of this arrangement could be obtained without greatly altering the normal-acceleration alleviation characteristics of the system by providing a small but reasonable static margin for the airplane. Interconnection of the flap-operating mechanism and the pilot's elevator control appears to allow satisfactory longitudinal control of the airplane. In addition, it was found that the interconnected system resulted in a more rapid change in normal acceleration than was obtained by elevator control alone.

Based on the results of the theoretical investigation just discussed a light transport airplane was modified at the Langley Laboratory to incorporate the controls necessary for gust alleviation. Fundamentally, the system used full-span wing flaps to maintain constant lift in flight through rough air. A portion of the elevator control was geared to the flaps to adjust the pitching moment due to flap deflection. An angle-of-attack vane located on a boom at the nose of the airplane was used to sense the variations in gust velocity. Signals from the vane were supplied to an automatic control system which operated the flap controls in proportion to the change in angle of attack. The automatic control system was designed to attenuate sharply the gust inputs at frequencies in the neighborhood of the airplane structural frequencies in order to prevent excitation of flutter.

The initial results, reported in Technical Note 3612, indicate that the system is at least capable of reducing the normal accelerations due to gusts by about 50 percent at a frequency of 0.6 cycle per second, the natural frequency of the airplane, and by about 40 percent at a frequency of 2 cycles per second. The controllability of the airplane with the gust-alleviation system in operation is adequate and it is felt that this type of control may result in improved handling qualities of the airplane.

As a result of the present interest in the spiral-stability problem associated with most personal-owner airplanes, the Flight Research Division of the Langley Laboratory has undertaken a program to investigate the effectiveness of a spiral-stability augmenting device. The specific problem facing the pilot of a personal-owner airplane is to maintain his airplane in wings-level flight during times when he has no natural-horizon reference and to keep the airplane from diverging spirally while he may be preoccupied with navigational problems. It has been demonstrated that the pilot's sense of orientation is unreliable in the absence of a visual reference, as may be the case when inadvertently or unavoidably encountering instrument weather. Also, many personal airplanes are equipped with only the basic instruments for instrument flight (turn indicator, ball-bank indicator, altimeter, and airspeed meter). Considerable proficiency in instrument flying is required to interpret the indications of these instruments properly and, in many cases, personal-airplane pilots are not sufficiently skilled in instrument flying to undertake it with safety. Although most present-day personal-owner airplanes, particularly those with high-wing designs, possess a slight degree of inherent spiral stability in cruising flight, they show unstable spiral tendencies under operational conditions. The main reasons for this apparent spiral instability are a lack of means for trimming the airplane laterally or directionally, a variation of lateral and directional trim with airspeed, and control-system friction which prevents the control surfaces from returning to trim position after a control deflection, even if there had been a means for initially trimming the airplane.

The device studied utilizes a rate-gyro sensing element to switch the control effort of an on-off type of control. This control deflects the ailerons at a constant rate relative to a spring preloaded aileron neutral position. An analytical study using phase-plane and and analog-computer methods was carried out to determine a desirable control-effort switching function for the on-off or nonlinear control used in this device. Results of the analytical study indicate that the nonlinear control is effective in compensating for directional trim changes of the airplane due to changing flight conditions and also in rapidly returning the airplane to level flight from an initial roll displacement. Results of the flight-test program reported in Technical Note 3637 essentially verified the analytical results; that is, the device is capable of maintaining the airplane in equilibrium over its operational speed range under directional out-of-trim conditions that would cause rapid divergence of the normal airplane, The device also prevents excessive heading wander and airplane gyrations in turbulent air without pilot control. A means for holding the airplane in a stabilized turn to facilitate mild maneuvering through use of the automatic control is provided.

As part of a general research program for testing various means of automatic stabilization, the Pilotless Aircraft Research Division of the Langley Aeronautical Laboratory has been conducting an investigation of various autopilot systems, including both linear and nonlinear systems. Nonlinear control systems have been considered as an effective, and possibly a simple way, of obtaining desired missile responses automatically. One such system, using a mechanical linkage with a dead or inoperative region as a means of providing an effective rate signal in an autopilot, has been studied in laboratory tests and through calculations of resulting missile motions in roll. Thus, a mechanical device is used to improve system performance in a manner similar to that which would be obtained by using a rate gyroscope. Laboratory tests of an actual servo using this device in connection with an electromechanical simulator of the rolling motion of a missile show the conditions under which a higher degree of stability is obtained with this nonlinear arrangement than with that of a comparable linear system. Results of this study are given in Technical Note 3602.

The Ames Aeronautical Laboratory has conducted extensive research on the various factors which influence a pilot's ability to track a target with fighter-type airplanes. Normally, simulated aerial attacks have been made against target airplanes and tracking errors have been evaluated from motion pictures of the target. In order to eliminate the trouble and expense of a second airplane, it appeared desirable to replace the target airplane by a simulated target projected on the windscreen. Selected attack situations and target maneuvers could thus be repeated quickly and accurately, and tracking errors could be continuously recorded for rapid analysis. To investigate this method, which is described in a paper presented at the Institute of Aeronautical Sciences, a prototype optical target simulator was constructed and evaluated in flight.[1] Initial experience with this equipment led to application of its principles to other flight-research devices and problems. The related equipment which has been developed includes a target simulator for tracking research on a scope-presentation fire-control system; an airborne drone aircraft simulator for use in studies of simple remote-control systems, and a windscreen-instrument display for the pilot's use in performing prescribed maneuvers.

Pilot opinions indicated that an optical target simulator (installed in an F–80 airplane) provided good qualitative matching of actual tracking problems. Tracking errors against actual and simulated targets were equal in steady, straight tail chases and slightly greater against the simulated target in steady turning flight. The most apparent difference was the large error against the simulated target during abrupt turn entries, due possibly to a lack of bank-angle information to the tracking pilot.

[1] See Kauffman paper listed on p. 77.

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The target simulator used with the scope-presentation fire-control system included a tracking radar for obtaining target-motion data, and an oscilloscope for displaying computed steering errors to the pilot for corrective action. It was the pilot's opinion that the scope-presentation target simulator gave a good representation of the steering problem encountered in final attack runs against actual targets.

In the airborne drone simulator, pilots' maneuvering commands to the drone are applied by use of a switch on the control stick. These signals are then fed to an airborne drone dynamic analog computer which approximates the behavior of the line of sight to the actual drone. The output of the analog then controls a dot appearing on the windscreen, indicating the simulated drone motions.

Principles and features similar to those used in these studies appear to have application to a wide variety of weapon-systems development, training, and instrument-flight problems.

INTERNAL FLOW

Inlets

Although external compression inlets have produced relatively efficient supersonic compression of the induction air, the wave drag of the external cowl has been high and as much as 10 to 20 percent of the total airplane drag at Mach numbers above 2. An experimental investigation was conducted at the Ames Laboratory therefore to study the feasibility of using inlets having internal contraction at supersonic speeds and in which the external angle of the lip would be small, thereby resulting in little or no drag penalty. The internal compression inlet studied was circular in cross section and had a translating center body which could be moved forward in order to increase the throat area of the inlet, thereby permitting supersonic flow to start in the converging portion of the inlet. The center body then could be moved rearward in order to reduce the throat area to nearly the isentropic value for a convergent-divergent nozzle. The results have shown that with such an inlet the external drag was essentially zero and the total pressure measured at the simulated compressor face was as good as that measured with single-cone, external compression inlets at Mach numbers from 2.1 to 3, the limit of the investigation.

The use of a deflector ahead of the inlet at angle of attack resulted in maintaining satisfactory pressure-recovery characteristics up to angles of attack of approximately 12°. Boundary-layer control on the center-body annulus through porous surfaces in the vicinity of the throat of the internal contraction inlet improved the pressure-recovery characteristics of the inlets by as much as 10 percent. In addition to satisfactory total pressure-recovery characteristics, the total pressure variation at the simulated compressor station was small and within 12 percent of the average value for the Mach number range from 2.1 to 2.5.

The design of the internal-flow system of a turbojet-powered aircraft for optimum operation at transonic or supersonic speeds usually compromises operation of the system during take-off and low-speed operation. In order to study this problem, two annular conical-shock inlets and several open-nose inlets of varying proportions and lip profiles have been investigated in the Langley 8-foot transonic tunnel under static operating conditions. Measurements were made of pressure recovery, lip pressure distribution, total-pressure distribution, and mass flow. Results indicate that the pressure recovery and the choking mass-flow rate were greatly improved by replacing the sharp lip on the open-nose inlet with more rounded lips. The open-nose inlets provided greater pressure recoveries than did the annular conical-shock inlets.

Charts and tables for the design of axisymmetric and two-dimensional inlets and exits have been prepared at the Lewis Laboratory and are presented in Technical Note 3589 for supersonic Mach numbers up to 4.0. The report also indicates a compression limit for isentropic inlets that restricts obtainable pressure recoveries to values considerably less than unity.

The internal space requirements for instrumentation and armament frequently dictate that the engine air inlet be located on the sides of the fuselage as a single- or twin-scoop configuration. An investigation has been conducted in the Langley 8-foot transonic tunnel of a twin divergent-walled submerged inlet. The pressure recoveries of the inlet at zero angle of attack varied from 98 percent at a free-stream Mach number, M_o, of 0.60 to 93 percent at $M_o=1.1$. Extensive total-pressure losses, together with severe flow oscillations, were observed at angles of attack greater than 7.3° for all Mach numbers.

Preliminary flight tests at the Ames Laboratory on a jet airplane equipped with submerged-type side inlets indicated performance considerably below the design estimates. Further flight tests were made on a similar airplane which featured a scoop-type inlet and a fuselage with a smaller aft end. In order to determine to what extent the inlet change contributed to the performance increase, the two different inlets were flight tested using the same after-fuselage configuration. The comparison was made on three bases: (1) The induction-system efficiency (ram recovery ratio at the inlet and at the compressor face, and engine power output), (2) the overall airplane drag coefficient, and (3) a computed factor of relative effectiveness.

The results indicate that the submerged inlet had a higher pressure recovery but also had a higher drag than the scoop inlet. Compared on the basis of a factor of relative effectiveness, the two inlet installations were found to be of about equal merit. Sealing the boundary-layer bleeds on the scoop inlet raised the low recovery (below a Mach number of 0.85), whereas sealing the bleeds on the submerged inlet decreased the airplane drag coefficient.

One of the problems associated with the design of an efficient twin-scoop air-induction system on the side of the fuselage is the manner of handling the boundary-layer air ahead of the inlet. Two methods of boundary-layer control in a twin-scoop induction system were recently investigated at the Ames Laboratory: one allowed the low-energy air to pass under a compression ramp placed one boundary-layer height away from the fuselage; and in the other a portion of the low-energy air was drawn off through a permeable compression ramp placed contiguous to the fuselage surface. The results indicated that in both cases the boundary-layer control had a favorable effect on the total-pressure recovery and inlet airflow steadiness. A comparison of the two types of boundary-layer-control systems investigated showed that, in general, the system in which the boundary-layer air was allowed to pass under the compression ramp had a higher net propulsive thrust and a larger stable range of operation than the system in which the boundary-layer air was drawn off through the permeable compression ramp.

In the design of auxiliary air inlets or boundary-layer-removal systems, estimates of the entering boundary-layer mass flow and momentum must be made. Charts have been prepared at the Lewis Laboratory for evaluating these quantities for various turbulent boundary-layer profiles and are presented in Technical Note 3583.

Inlets designed for supersonic jet engines are generally unstable at certain operating conditions. It is desirable to determine criteria which define the stable operating range of such inlets. This problem has been studied at the Lewis Laboratory and is presented in Technical Note 3506, wherein the jet engine is likened to a Helmholtz resonator having a through flow. It is assumed therein that at each instant, conditions in the combustion chamber are uniform. The latter assumption is relaxed in Technical Note 3574, which applies acoustic impedance techniques in a study of inlet stability.

Supersonic inlets of ram-jet-engine installations are also susceptible to unstable oscillations when the engine is not operating at either design Mach number or air-fuel ratio. Since this unstable inlet operation, known as buzz, has a strong detrimental effect on ram-jet performance, it is essential that this phenomenon be thoroughly understood so that the problem may be eliminated. Technical Note 3695, prepared at the Langley Laboratory, reports on a study, in which a modified nonstationary one-dimensional wave theory is employed, of the various waves which comprise the buzzing cycle. This analysis predicts the complex pressure-time variation in the ram jet, found experimentally, and also shows how the oscillation begins and is perpetuated.

Diffusers

Tests have been conducted with a series of diffuser shapes defined by empirical equations having constants which are related to the initial boundary-layer thickness. The shapes were designed to provide high pressure recovery at near-sonic inlet Mach numbers. The tests were conducted at the Ames Laboratory through the mass-flow range with a variety of attached and separated initial boundary layers and also with two offset ducts, with a variety of surface conditions in the shortest duct.

The experimental results, reported in Technical Note 3668, provide substantiation for the design trends upon which the empirical equations were based. In addition, it was found that, for the axially symmetric ducts tested, the effect of initial boundary-layer thickness on pressure recovery was as important as that of duct shape. The best performance was obtained with a short duct with a thin initial boundary layer. With separated boundary layers, extended entry lengths provided markedly improved pressure recovery and flow steadiness relative to a similar duct with no entry extension. The offset ducts suffered losses in all performance parameters relative to a similar axially symmetric duct. Near maximum mass flow, the surface conditions investigated had only small adverse effects on pressure recovery. A loss of several percent in total-pressure recovery occurred with air leakage into the duct near the throat.

In many installations a short diffuser would be extremely desirable, but its use has been prevented by the total-pressure and static-pressure losses incurred by airflow separation resulting from the large included angles of a short diffuser. To determine whether this separation could be eliminated by using area suction, an exploratory investigation was performed with a 30° and a 50° porous, conical diffuser having area ratios of 2. The tests, at the Ames Laboratory, made at a mean inlet Mach number of 0.2, indicated that area suction eliminated airflow separation in the 30° and 50° diffusers with suction mass flows of 3 and 4-percent of the inlet mass flows, respectively. By eliminating airflow separation, the total-pressure and static-pressure losses of the 30° and 50° diffusers were less than those of a 10° diffuser without boundary-layer control.

Studies of various means of boundary-layer control for the purpose of increasing the efficiency of short diffusers have continued at the Langley Laboratory also. The results obtained in two investigations of the application of vortex generators, boundary-layer suction, and high-energy-air injection into the boundary layer to short, annular diffusers show that vortex generators are ineffective for extremely short diffusers. It also was found that a net gain in overall efficiency cannot be obtained by suction or injection boundary-layer controls unless the auxiliary pumping power is held to a minimum by eliminating all flow separation upstream of the control point and by specially shaping the wall contours downstream of this point to suit the flow conditions prevailing with the control operating. The increase in static-pressure rise obtained through the use of boundary-layer controls was of the order of 40 percent.

Ducts and Outlets

In continuation of the studies of the Langley Laboratory of means for increasing the efficiency of internal-flow systems, an investigation was conducted to determine the maximum flow capacity and the pressure drop of 90° bends in ducts of circular cross section. It was found, in Technical Note 3696, that a mean radius of turn of 2.5 duct diameters was approximately optimum with regard to both characteristics. A duct bend with this radius of turn provided a maximum flow equal to 95 percent of the theoretical choking flow and experienced an increase in total-pressure-loss coefficient of only 31 percent as its entrance Mach number was varied from low values to the choking value. The effects of changes in inlet boundary-layer thickness were found to be unimportant.

The high auxiliary-air requirements of high-speed aircraft continue to emphasize the importance of the handling of this air. In order to obtain additional design information applicable to this problem, a transonic investigation of the performance of rectangular thin-plate and ducted outlets located in a surface parallel to the flow has been made by the Langley internal aerodynamics laboratory. As reported in Technical Note 3466, it was found that the thin-plate outlets provided higher discharge coefficients at low discharge pressure differentials and that their performance increased substantially as their aspect ratio was decreased below unity. All of the flush ducted outlets had approximately the same discharge coefficients regardless of inclination or curvature of the duct axis and provided more flow than the thin-plate outlets at high supply pressures. The discharge coefficients of these outlets were increased by recessing their downstream surfaces. Surface dynamometer measurement showed that the thrusts produced by outlets of these types corresponded approximately to the streamwise components of the calculated theoretical thrusts.

Thrust Reversers

Many uses for thrust reversers on jet aircraft have been proposed. They include braking the landing roll, reversing or spoiling thrust during the landing approach so that maximum engine speed may be maintained, and braking during diving maneuvers to limit flight speed. To be used effectively, the reverser must give the desired amount of reverse thrust without affecting engine operation. Also, the design must lend itself to stowage with a minimum amount of boattail or base drag.

As part of an overall investigation of thrust reversers and their associated problems, studies were conducted with cold flow on thrust-reverser models. This work was done on a small-scale unheated-air-duct setup equipped with a 4-inch-diameter exhaust nozzle at the Lewis Laboratory.

The performance of a hemispherical thrust reverser over a range of geometric variables and some of the factors that affect reverse-thrust performance were investigated. The effects of most of the design variables of the reverser were obtained at an exhaust-nozzle total- to ambient-pressure ratio of 2.0. The basic data over a wide range of conditions were also obtained, however, so that other comparisons could be made.

The performance of cylindrical-type thrust reversers and the effects of several modifications on their performance were also investigated at an exhaust-nozzle pressure ratio of 2.0. These modifications included changes in frontal area, width-to-height ratio, depth, lip angle, end-plate depth, and end-plate shape. The performance of swept-type cylindrical thrust reversers, the relation of reverse-thrust ratio to reversed-flow attachment, and thrust-modulation characteristics were included in the study.

Preliminary data have been obtained on the performance of several cascade-type thrust reversers located upstream of the exhaust nozzle at exhaust-nozzle pressure ratios up to 2.4. Such reversers are referred to as the tail-pipe-cascade type. A total of 15 different tail-pipe-cascade configurations was investigated. These included two blade shapes, several cascade length-to-span ratios, and various innerbody lengths. For some of the configurations, modulation performance and surveys of total pressure and flow angle were obtained at the cascade discharge.

Technical Note 3664 presents several types of thrust reversers investigated under the overall NACA program, summarizes the important performance characteristics, and presents proposed operation methods. Three types of reversers were investigated: target, tail-pipe cascade, and ring cascade. The effects of design variables on performance, reversed-flow fields, and thrust-modulation characteristics were determined for each type.

PROPELLERS FOR AIRCRAFT

Aerodynamic Problems

Continued interest in propellers as propulsive devices capable of efficiently utilizing the high power available from modern gas-turbine engines has emphasized the need for data defining their characteristics in their various flight regimes. In addition to their performance at high speeds, there is equal need to know their behavior in the negative-thrust range, where extremely large forces can result in the event of engine failure and in the high-thrust conditions occurring during the low-speed take-off run. The dual-rotation propeller is of particular interest since it offers the advantage of smaller diameter, higher efficiency, absence of reaction torque, and less noise as compared with the single-rotation propeller capable of absorbing the same power. To provide data useful in the design and development of these propellers, an investigation has been made in the Ames 12-foot pressure wind tunnel of NACA 4–(5)(05)–037, six- and eight-blade, dual-rotation propellers. These propellers were operated at positive and negative thrusts, at blade angles from $-20°$ to $70°$, and at Mach numbers up to 0.90, including operation at low-speed, near-static conditions. At high speed, M=0.72, these propellers had an efficiency of 0.80 for a blade angle of 65°, and at low-speed near-static conditions they produced about 3 pounds of thrust per horsepower at their design power loading.

The flow of air through the plane of a propeller is greatly influenced by the type of spinner used. Generally, propellers are designed for operation in a flow having uniform axial velocity over the propeller disk. Frequently this ideal flow is not realized because of spinner disturbance to the flow field. A flight investigation was made on three full-scale propeller spinners using the XF–88B propeller research airplane to obtain information on the spinner pressure distributions and on the radial extent of flow-field distortion by the spinner. One was a conventional conical spinner and the others, differing in size, had spherical midsections. Measurements reported in Technical Note 3535 show that distortion of the flow in the propeller plane amounting to 10 percent of the free-stream velocity extended outward from the spinner surface from 30 to 100 percent of the spinner radius. On the large spherical and conical spinner, the extent was greater than with the small spherical spinner; this indicates that there are some shapes for which the extent of flow disturbance can be minimized. Also, it is apparent from measurements with the conical spinner that the velocity of flow may be considerably reduced in the vicinity of the spinner. It is also pointed out that unfavorable flow separation could result with spherical spinners.

As part of a general investigation of the static characteristics of propellers, empirical methods for estimating static thrust, based on propeller charts, have been compared with a newer strip-theory method. The comparisons of experimental data with thrust and power coefficients calculated by strip theory show good agreement where adequate airfoil data are available, and the strip theory yields better results than the empirical method.

In the increasingly important field of vertical and short takeoff and landing, high thrust at low speeds and cancellation of torque reaction make dual-rotation propellers an attractive means of propulsion. In order to fulfill a need for information concerning dual-rotation-propeller static-thrust characteristics, an investigation of six- and eight-blade dual-rotation propellers operating at zero advance was conducted at the propeller static test stand of the Langley 16-foot transonic tunnel. These propellers were made up of blades having the NACA 8.75–(5)(05)–037 design number. Static thrust and torque characteristics were investigated. Some types of vertical-take-off aircraft depend, for longitudinal and directional control, at zero or low forward speeds, upon the action of the slipstream on control surfaces. Velocity distributions, therefore, were also measured in the wake of the six- and eight-blade dual-rotation propellers. A major part of the investigation of dual-rotation propellers pertained to a study of vibratory stresses excited by mutual interference of the propeller blades of the front and rear components. The principal finding is that blade passage in a dual-rotation propeller is a fairly strong source of aerodynamic vibratory excitation. High stresses, however, were not encountered under any of the test conditions, and only moderate stress rise was encountered at resonant conditions, primarily in the torsional mode.

In connection with research leading to improved aerodynamic qualities in vertical-take-off aircraft, an investigation (reported in Technical Note 3547) was made to determine the effects of airspeed and angle of attack on the lift, drag, and pitching moment of a shrouded-propeller model. The model had a shroud length of about two-thirds the propeller diameter and was tested over an angle-of-attack range from 0° to 90°. Tests were made of the complete model with the propeller operating and also of the shroud and motor combination with the propeller removed. Static-thrust efficiency was found to increase with increasing radius of the shroud inlet lip.

Propeller Stall Flutter

In the design of a supersonic-type propeller, primary consideration has been given to reducing blade-section thickness ratios to as low a value as stress considerations will permit. However, the propeller blade with very thin sections has low torsional rigidity and is susceptible to stall flutter; consequently, the power capacity of such propellers during take-off may be limited by flutter rather than by aerodynamic characteristics. In an effort to improve the flutter characteristics of propellers and yet maintain thin blade sections, the effect of blade-section camber on the stall-flutter characteristics of three NACA propellers was investigated. An increase in blade-section camber was found to produce an important reduction in blade stresses due to flutter at blade-angle settings up to 28°.

HELICOPTERS

Rotor Aerodynamics

The need for research work on helicopters arises both from the desire of the user for a better, more reliable, or less expensive helicopter to do an existing job and from the desire to develop designs which will permit successful application to new missions. For both purposes, improved rotor aerodynamic theory and coordinated experimental research are required. In recent years, modern high-speed automatic computing machines have become generally available to research institutions and to industry. This availability, in turn, has made possible the application of these machines to the problem of computing the aerodynamic characteristics of lifting rotors by numerical methods. By means of such methods, factors that are normally omitted from conventional analytical rotor treatments, such as stall and compressibility effects and combinations of such design parameters as hinge offset, blade twist and taper, and root cutout, can be accounted for. Greater accuracy is thus obtained for designs of conventional types, while design studies for radically different helicopters (particularly designs aimed at higher forward speeds) become practical. The necessary equations and procedures for carrying out the numerical computations involved are presented in Technical Note 3747. Rotor characteristics considered included thrust, profile drag, total power, flapping, rolling and pitching moments, direction of the resultant-force vector, and the harmonic contributions of the shear-force input to the hub.

A knowledge of the steady-state flapping behavior of lifting rotors is necessary in the design of helicopter hubs and control systems, in estimating rotor-fuselage clearances, and as a prerequisite to the numerical evaluation of the aerodynamic characteristics of rotors. Equations for calculating rotor-blade flapping have been available from various theories. In order to reduce computation to a minimum, theoretical flapping values can be obtained directly from charts which were constructed and are presented in Technical Note 3616. The charts are applicable over a wide range of helicopter operating conditions and for blade twists of 0°, −8°, and −16°.

The recent emphasis on helicopter-rotor vibration and the consideration of compound helicopters and convertiplanes require a more complete evaluation of the induced velocity field near a lifting rotor and of the rotor downwash in the regions of wings and tail planes. The preliminary report of this experimental investigation has been supplemented in Technical Note 3691 by a complete presentation of results and a comparison with theoretically predicted flow fields.

The normal component of induced velocity of a lifting rotor can be calculated with good accuracy as far rearward as three-quarters of a diameter behind the front edge of the rotor, provided a realistic nonuniform (essentially triangular) disk loading is assumed. Because of the rapid rolling up of the trailing vortex system, the induced-velocity calculations at the rear quarter of the disk do not accurately predict the measured results. In the far field behind the rotor the induced flow can be more accurately predicted by considering a uniformly loaded rotor in the same manner as a rectangular wing. Charts of the normal component of induced velocity in the longitudinal plane of symmetry in the near and far fields of the rotor determined analytically for different nonuniform, circularly symmetrical disk loadings are presented in Technical Note 3690.

One of the more difficult problems facing the helicopter designer is the calculation of the rotor-blade aerodynamic loading. The problem is difficult because of the complexity of the rotor flow and the uncertainties of the assumed distribution of induced velocity across the rotor disk. An extensive investigation to determine experimentally the helicopter rotor-blade loads for both the hovering and forward-flight conditions has been conducted in the Langley full-scale tunnel. The static-thrust information on a single rotor is presented in Technical Note 3688. A rapid dropoff in load per foot of span is shown near the blade tip. A comparison of the blade-section loadings with theory shows that a tip-loss factor which varies with thrust coefficient gives more accurate results than the commonly employed constant tip-loss factor.

The effects of compressibility arising from high tip-speed operation on the flapping, thrust, and power of a helicopter rotor over a wide range of forward flight conditions were investigated by the use of numerical methods and are reported in Technical Note 3798. With the particular airfoil characteristics used, the results indicated minor increases in rotor flapping and thrust when rotor tip speed was increased from 350 to 750 feet per second. The largest effect noted was an increase in profile-drag power in the advancing side of the disk that was proportional to the amount by which the blade-tip Mach number exceeded the drag-divergence Mach number. These effects of compressibility appeared to be independent of blade twist but are, of course, a function of the airfoil characteristics employed in the analysis.

Propulsion

The large load-lifter type of helicopters are particularly well suited to benefit from use of rotor-blade tip-mounted jet-propulsion power plants. A pulse-jet unit having high ratios of thrust to frontal area had previously been studied on the Langley helicopter test tower. In order to determine centrifugal-field effects on the propulsive characteristics, the same pulse-jet unit was tested over a range of yaw angles and forward speeds in the Langley 16-foot transonic tunnel. A comparison of the nonwhirling and whirling results, presented in Technical Note 3625, indicates that the pulse jet is subject to reduced performance. This reduction results from centrifugal distortion of the fuel-spray pattern for centrifugal accelerations greater than 200g (a value high enough to indicate that no difficulty of this nature should be encountered with the largest load lifters but indicating an area requiring careful consideration for smaller pulse-jet designs).

A performance analysis of fixed- and free-turbine helicopter engines has resulted in performance charts and comparisons of the off-design specific fuel consumption, altitude performance, power-speed characteristics, and response times. The results presented in Technical Note 3654 indicate that power modulation of the fixed-turbine engine was more rapid than the free-turbine engine at constant shaft speed, although simultaneous changes in speed and power were executed by both engines in about the same time. At constant temperatures, the free-turbine power varies only slightly with shaft speed, whereas the fixed-turbine power decreases significantly with shaft speed.

Solutions of the problems of excessive vibration, structural fatigue, roughness of control, and rotor interference would become more evident if the nature of the rotor disturbances was known. With a reasonable knowledge of inflow variations, it may at least be possible to design away from these adverse characteristics. The available current experimental inflow data are not adequate to permit a thorough evaluation of existing theories. With the exception of the hovering condition, therefore, only a limited amount of material has been published about the correlation between inflow theory and experiment. Since no force and moment data for offset-flapping-hinge rotors were available, a study was undertaken by the Massachusetts Institute of Technology, under the sponsorship of the NACA.

Inflow distributions, azimuth and spanwise, were determined analytically from measured pressure distributions and blade-motion data on a model helicopter rotor blade under hovering and simulated forward-flight conditions. Pressures and corresponding blade flapping were recorded for various rotor conditions at tip-speed ratios of 0.10 to 1.00. Covered in this study are one-bladed-rotor operation effects, deliberate blade stall, data on the effects of cyclic pitch, and tests on a rotor with a 13-percent-offset flapping hinge. Since the offset-flapping-hinge rotor was used primarily as a means of alleviating stall in order to obtain inflow data at high tip-speed ratios, μ, in the vicinity of 1.0, no cyclic pitch was used to balance out the hub moments resulting from the incorporation of offset hinges. It is these moments which are the primary source of stall alleviation. The inflow plots presented in Technical Note 3492 indicate variations very different from the uniform distributions which are sometimes associated with a rotor disk. An extensive investigation of the $\mu=0.30$, zero-offset rotor condition showed that larger inflow variations than those predicted by theory can exist. In addition, however, upflow over the forward portion of the disk and relatively large induced velocity at the trailing edge are verified. The inflow patterns for the zero-offset and 13-percent-offset rotors under the same conditions of operation, except for the presence of hub moments in the offset-hinge case, are found to be very different in general character.

SEAPLANES

Hydrodynamic Elements

Experimental and theoretical research on planing surfaces has been extended to include pressure-distribution surveys for a series or related prismatic planing surfaces having angles of dead rise from 0° to 40°, with and without chine flare. These pressure distributions are presented in Technical Note 3477 for a wide range of wetted length and trim.

The results substantiate the use of the normal-load coefficient as the key parameter in predicting flat-plate center-line pressures. The results further show that flat-plate pressure distributions can be adequately predicted from existing theories. The reduction in pressure accompanying an increase in angle of dead rise is about as expected on the basis of previous force measurements. The addition of horizontal chine flare increases the pressure near the chines and extends the region of positive pressures further forward of the stagnation point in the vicinity of the chines. Existing theories are in poor agreement with the experimental pressure distributions obtained for surfaces having dead rise. The lift and centers of pressure, predicted on the basis of the pressure distributions, are in good agreement with recent experimental and theoretical NACA research on planing surfaces.

Interest has been developing in the operation of water-based aircraft off ramps or beaches where the water depth approaches zero. In view of this, an experimental investigation was made to determine the effect of shallow water on the hydrodynamic characteristics of a flat-bottom planing surface. These data are reported in Technical Note 3642 and show that the lift, drag, and trimming moment about the trailing edge of the model increased as the clearance between the model and the tank bottom decreased. The most apparent increases occurred at clearances below one beam. With combinations of high-wetted length and high trim, however, the values began to increase at somewhat greater clearances. The lift-drag ratio increased with decreasing clearance for wetted length-beam ratios greater than 0.8 and trims less than 16°. The roach in the wake of the model increased in height and moved aft of the model as the clearance decreased.

In the past, seaplane-spray investigations were primarily concerned with the definition and reduction of spray impinging on the seaplane. Recent developments have somewhat altered the spray considerations since modern seaplane designs have closely coupled aerodynamic and hydrodynamic components which are constructed strong enough that considerable forces may be developed on the surfaces by impinging spray. A study of the scale relations for converting model spray-force data to full size is reported in Technical Note 3615. The results show that spray lift forces can be scaled by the conventional Foude relations but that a Reynolds number effect on spray drag is indicated. An empirical method is suggested for correcting the spray frictional-drag coefficients on a Reynolds number basis.

Results of a preliminary investigation of self-excited vibrations of a single planing surface are reported in Technical Note 3698. Research on vibrations of planing surfaces is of considerable significance in the application of hydro-skis to water-based aircraft, since such vibrations have been known to cause structural damage to the aircraft. This research has indicated that self-excited vibrations occur with high aspect ratio (on the order of 10) of the wetted portion of the planing surface and appear to be essentially an oscillation in trim or rise, or a combination of these motions. The oscillations can be decreased in severity or eliminated by using planing surfaces which limit the wetted aspect ratio. Dead rise, transverse curvature, and a pointed trailing edge are all effective.

In order to provide for flush retraction of hydro-skis on high-speed water-based aircraft, it is sometimes desirable to form these components from portions of the airplane which can be extended for landing and take-off. Since the bottom of these skis will then conform to the shape of the fuselage which is generally rounded or to that of the wing which is more or less flat, the skis also will generally have rounded or flat cross sections. Because of this, an investigation was initiated to determine the characteristics of planing surfaces of several plan forms and transversely curved bottoms. One surface was of rectangular plan form with a flat bottom; the second had a rectangular plan form with transversely curved bottom; and a third surface had a flat bottom but was triangular in plan form. The trims investigated ranged from 4° to 20°. The data were reduced in the form of load, resistance, trimming moment, and draft plotted against wetted area.

Research Equipment and Techniques

The rapid increase in the landing speeds of current airplanes has caused a corresponding increase in the water speeds at which seaplanes operate. As a result, the gap between the speeds available in the existing hydrodynamic testing facilities and full-scale speeds has widened to an extent sufficient to make it advisable to ascertain whether these differences in speed are causing any significant differences in force coefficients. In an attempt to close this gap, an investigation has been made of the feasibility of obtaining hydrodynamic data at full-scale speeds by utilizing a rectangular 3- by 3¼-inch free-water jet actuated by compressed air. A comparison of planing data obtained in the water jet with similar data obtained in conventional towing tanks indicates that it is feasible to use a free-water jet as a hydrodynamic test facility for obtaining planing data at very high speeds. The main problem appears to be in establishing an adequate method of correcting the jet data for the limited boundaries. Consideration has been given to a simple empirical method of correcting planing data for the jet boundaries. This method gave reasonable results for the limited data available.

POWER PLANTS FOR AIRCRAFT

The ever increasing range, speed, and altitude requirements specified for supersonic airplanes and missiles has necessitated consideration of new types of power plants as well as the refinement of the more familiar types of turbojet, ram-jet, and rocket engines. As in the past the goal of NACA research in this field has been information leading to the development of light-weight, highly efficient propulsion systems, utilizing both chemical and nuclear fuels.

Increased emphasis has been placed on new fuels and rockets problems and on the effects of high temperature environment on engine performance and reliability. Of particular interest at high altitude is the performance of the combustion chamber. Much research has been done in order to understand the differences between theory and experiment for the effects of turbulence on flame ignition, propagation, and heat release.

In order to summarize the present "state-of-the-art" of airbreathing powerplants research, a technical conference was held at the Lewis Laboratory in December 1955.

A description of the Committee's recent unclassified research in the field of aircraft propulsion is given on the following pages.

AIRCRAFT FUELS

Fuels Performance Evaluation

Because the ram-jet engine requires no significant moving parts for its operation, a greater variety of potential fuels is available for the ram-jet engine than for the turbojet and reciprocating engines. Specifically, the ram-jet engine is capable of utilizing fuels that may produce considerable solid materials in the process of combustion. A number of conventional and unconventional ram-jet fuels have been suggested for research because of the need for ram-jet fuels that may permit the realization of flight range and thrust beyond the limits attainable with conventional hydrocarbon fuels. The adiabatic combustion flame temperature, combustion equilibrium-gas composition, air specific impulse, and fuel-weight specific impulse were investigated for each of several fuels. These fuels included octene-1, aluminum, aluminum-octene-1 slurries, magnesium, magnesium-octene-1 slurries, diborane, pentaborane, boron, boron-octene-1 slurries, hydrogen, α-methylnapthalene, and graphite.

The desire for higher flight speeds has increased the demand for greater power per pound of fuel, and per unit volume of fuel. Emphasis has therefore been placed on the development of ram-jet engines and on turbojet engines equipped with afterburners suitable for operation on special fuels.

An analytical investigation was conducted to determine the theoretical air-specific-impulse performance of several fuels over a range of equivalence ratios, inlet-air temperatures, and combustion pressures. The fuels included octene-1, 50-percent-magnesium slurry, boron, pentaborane, diborane, hydrogen, carbon, and aluminum. Inlet-air temperatures between 100° and 900° F are considered at a combustion pressure of 2 atmospheres; a combustion pressure of 0.2 atmosphere is also considered at an inlet-air temperature of 100° F. Also discussed are the determination of air-specific-impulse efficiency and combustion efficiency for experimental data by the use of theoretical results and the estimation of the relative amounts of various fuels required to maintain a fixed engine thrust level.

For some time fuel cost has been accepted as a major consideration in the conversion of commercial airline operations from piston-engine aircraft to gas-turbine-engine aircraft. The most frequently proposed fuel for use in gas-turbine transports is kerosene; however, there has been interest in the possibility of further fuel-cost reductions by use of low-cost distillate and the residual fuel oils. These fuel oils are considerably cheaper than aviation gasoline, and the residual types are substantially less costly than kerosene. Despite the cost attractiveness the physical properties and combustion characteristics of such fuels offer many problems that must be solved before they can be successfully utilized in commercial aircraft. The properties of distillate and residual fuel oils and the influence of these properties on engine performance and handling procedures have been studied.

Combustion-chamber carbon deposition is a significant factor influencing the choice of fuels for the turbojet engine. A number of carbon-deposition tests were conducted in a single-tube combustor and in full-scale engines to provide some information concerning the relation between single-combustor and full-scale-engine carbon deposition. These data together with limited flight operational data provide some indication of the tolerable limits of deposition and of the necessary control test limits that would ensure adequate fuel quality.

Thermal Properties of Aircraft Fuels

Sodium, beryllium, and magnesium alkyls thermally decompose under vacuum at 100° to 200° F. The main products are a metal hydride or an alkyl metal hydride and an olefin of the same number of carbon atoms as the original alkyl group. Recently, it has been observed that a boron alkyl, tri-n-butylborane, decomposed upon heating under vacuum; the products were dibutyldiborane and butene.[2] The similarity both in the nature of the products and the experimental conditions suggests that the same mechanism is involved in all these metal alkyl decompositions.

In some present-day turbojet engines, the engine lubricant is cooled in a heat exchanger by means of the fuel as it flows to the combustor. As a result, the fuel may be heated to temperatures high enough to cause the formation of insoluble, gum-like substances. These insoluble materials, if not removed or adequately dispersed, may foul the heat exchanger so that the lubricant is not adequately cooled or clog the filter screens or the injector orifices so that the fuel flow is reduced to the point of engine failure. The thermal stability of jet fuels is affected not only by the temperature, but also by the residence time of the fuel in the heated zone. Since fuel-flow rates decrease with increasing altitude, the problem becomes more severe at high altitudes where both higher fuel temperatures and longer residence times are encountered. The effect of dissolved oxygen on the filter-clogging characteristics of three JP–4 and two JP–5 fuels was studied at 300° to 400° F. in a bench-scale rig employing filter paper as the filter medium.

Synthesis and Analysis

Only recently have accurate thermochemical data on boron-containing molecules become available. The data have been used to calculate the electronegativity of boron and the boron-to-boron bond energy which were used to compute the average bond energies $\overline{D}_{gm}(B–Z)$ from the Pauling electronegativity equation. The calculated values of $\overline{D}_{gm}(B–Z)$ were compared with experimental values, and the differences between the values have been interpreted in terms of partial double-bond character and resonance in BZ_3 compounds.

Recent measurements of the lattice constants of the alkali metal borohydrides make possible a theoretical calculation of the lattice energies of $NaBH_4$, KBH_4, $RbBH_4$, and $CsBH_4$. The entropies and free energies of the reactions of BH_3(g) and B_2H_6(g) with H^-(g) to form BH_4^-(g) can also be calculated. These various thermodynamic properties were obtained in an investigation.[3]

[2] See paper by Rosenblum listed on p. 78.
[3] See paper by Altschuller listed on p. 76.

COMBUSTION

Fundamentals of Combustion

Although conditions which may lead to the spontaneous ignition of fuels have been a subject of investigation for many years, the results of various researchers are in most instances not directly comparable, since the experimental conditions at the point of ignition may vary. An attempt was made to clarify in particular the effects of such parameters as fuel and oxygen concentration, pressure, temperature, and inert diluents on ignition delay. Furthermore, such data may help to elucidate the chemistry of the ignition reactions. Reactants were mixed at the desired temperature in a time much shorter than the ignition lag. The results on propane flames were obtained over a wide range of fuel percentages at atmospheric pressure and temperatures from 520° C to 740° C.

The spontaneous ignition temperatures of eight structurally different hydrocarbons were determined and correlated with the behavior of the same hydrocarbons toward vapor-phase oxidation (Technical Note 3579).

Much work has been done on the relative ease of oxidation of hydrocarbons, but the similarity in susceptibility to oxidation of the different bonds within the hydrocarbon molecule makes it difficult to isolate the separate steps in the reaction. The introduction of a metal atom into a molecule may add certain bonds which are more easily attacked by oxygen than the remainder of the molecule, and, hence, it may be possible to follow more readily the initial attack on the molecule by oxygen. Because of its similarity to carbon, silicon is a reasonable metal atom to use in such a study. Consequently, several alkylsilanes were prepared, and a study of the oxidation properties of these fuels was conducted.

In spite of the expanded efforts on combustion research in recent years, there is relatively little information on the detailed mechanism of the combustion of metals. The gross physical mechanism of the combustion of relatively small quantities of magnesium metal was studied in order to determine whether a vapor phase or surface mechanism is involved; further, to examine the effect of the composition of the oxidizing atmosphere, both as a matter of general interest and to determine if such studies might be informative as to the details of the physical and/or chemical processes involved.

A previous investigation resulted in a theory of spark ignition in nonturbulent and turbulent flowing gases using long-duration discharges. This theory is based on thermal processes and relates the spark discharge energy with gas density and velocity, electrode spacing, spark duration, intensity of turbulence, and constants of the fuel. A recent study was conducted to show the effect of fuel-air ratio and initial temperature on spark-ignition energy and to apply these results to this developed theory of ignition. Data were obtained at a pressure of 5.0 inches of mercury absolute, gas velocity of 50 feet per second, low-turbulent flow condition, and with long-duration spark discharges.

Flame quenching may be studied by simple experiments. A flame is introduced into one end of a tube filled with a combustible mixture. The flame will either burn through the length of the tube or will be extinguished (quenched) by the tube. Experiments show that the ability of a flame to get through the tube depends on the following factors: (1) pressure; (2) temperature; (3) kind of fuel, oxidant, and inert diluent; (4) relative concentrations of fuel, oxidant, and inert diluent; and (5) cross-sectional shape and size of the tube. Recently, a diffusional quenching equation was proposed. This equation correlates the effect on quenching of such diverse factors as fuel type, tube geometry, fuel and oxygen concentrations, and temperature. A thermal analog of this equation has been developed and shown to correlate the same quenching data satisfactorily. The two equations predict effects of different extent when helium is replaced by argon in an inflammable mixture. Thus, comparisons with experiment should indicate which of the two approaches is more useful. Consequently, a study was made of the effect produced by replacement of helium by argon on the quenching of propane-oxygen-inert flames.[4]

Recent flame-quenching research has indicated that there should be a set of simple relations among the various channel geometries which are capable of just quenching a given flame at a given pressure. In an investigation recently conducted and reported, the following points were discussed: (1) The derivation of a set of equations which predict the relations among the dimensions of a number of simple geometries capable of just quenching a given flame at a given pressure and (2) the evaluation of several of these relations by determining experimentally the wall quenching of downward-propagating propane-air flames as a function of fuel-air ratio and pressure for rectangular slots, cylinders, and cylindrical annuli.[5]

With the growing interest in high-speed combustion systems in which the attainment of greater heat-release rates per unit volume is demanded, it becomes more important to know how faster-burning fuel-air mixtures can be obtained. It has long been known that one way to increase the burning rate of a gaseous mixture is to preheat it. The question arises as to how far this process of increasing the flame velocity by increasing the initial temperature can be carried. Therefore, a study was made to (1) determine experimentally whether the flame velocity is affected by contact time between fuel and air at temperatures previously studied, (2) extend propane-air flame velocity data to higher

[4] See paper by Potter and Berlad listed on p. 78.

[5] See paper by Berlad and Potter listed on p. 76.

temperatures, (3) increase contact times and temperatures until appreciable effects of slow oxidation on flame velocity were observed, and (4) obtain information on the major products of the slow oxidation in near-stoichiometric propane-air mixtures as determined under steady-state flow conditions.

Space-heating-rate measurements can yield information useful in characterizing turbulent combustion. Technical Note 3277 indicates that for Bunsen burner flames stabilized over a field of pipe-induced turbulence, the space-heating rate was decreased with increasing linear flow and burner diameter and was independent of pilot conditions. Turbulent burning velocities over the same flow range were correlated by linear velocity at constant burner diameter, but the variation with burner diameter could not be expressed by a simple correlation.

For laminar flames, the determination of burning velocity is fairly direct because the reaction zone is thin and stream tubes may be identified. Turbulent flames, however, are not so easily characterized in this way. Even when a turbulent flame is stabilized, the volume occupied by the fluctuating reaction zone is large compared with the volume of a laminar flame zone. A characterization parameter, the space rate, for turbulent flames has been defined.[6] Measurements show that the space rate is directly proportional to the laminar burning velocity and indirectly proportional to the burner diameter. The space rate multiplied by the heat of combustion gives directly the heat-release rate per unit volume of space needed for the combustion.

The effect of turbulence on the propagation of a free flame has recently been investigated and reported. In this study free flames were spark ignited in a flowing turbulent stream of combustible gas. The propagation rate of the free-flame globule was observed photographically and with ionization gaps which defined the cone swept out by the globule as it was carried along by the stream.

Jet engine requirements have increased the need for understanding the mechanism governing the high volumetric heat-release rates of a fuel-air mixture. The burning velocities of premixed open propane turbulent flames and the effect of turbulence and turbulent-flow parameters on a variety of these flames was studied (Technical Note 3575).

The turbulent mixing of mass and heat is of major importance to the performance of jet-engine combustors. For example, mixing is largely responsible for the preparation of desired fuel-air mixtures and also affects the overall combustion process. In Technical Note 3570 an experimental comparison is presented between the Lagrangian and Eulerian correlation coefficients in the homogeneous isotropic central core of a turbulent pipe flow. The Lagrangian correlation coefficient was characterized by measurements of the turbulent diffusion of helium from a point source. The general objective of this investigation was to determine relations between turbulent-mixing theory and experiment that would provide sufficient information for the direct solution of practical mixing problems.

Low-volatility fuels, such as JP–5, are desirable for use in high-altitude jet engines from the standpoint of low fuel-tank losses. Also they present less fire hazard, under some conditions, than currently used JP–4 fuel. The very low evaporation rate of such fuels, however, may yield poor combustor performance. Therefore, knowledge of the factors that affect the evaporation rate of JP–5 fuel is useful to the combustor designer. A continuous sampling-probe technique was used to determine the percentage of JP–5 fuel spray evaporated under conditions common in ram-jet engines.

Experimental and calculated mass and temperature histories of fuel drops vaporizing with a constant velocity relative to the air were determined and reported in Technical Note 3490. Under some conditions the unsteady state or time required for the drops to reach the wet-bulb temperature is an appreciable portion of the total vaporization time.

The burning of clouds of particles, such as those existing in fuel sprays, has received relatively little attention because of the difficulty of the problems. With respect to theory, there are such complicating factors as interaction between drops, extinction of flame surrounding drops at high relative drop-air velocities, and complex turbulent systems such as those existing in the combustion chamber. There may be conditions under which diffusion is no longer the controlling element, and the rate of chemical reaction becomes dominant. A discussion of those factors normally associated with diffusion flames is presented. In addition, the efficiency of combustion of liquid fuels in turbojet combustors as affected by fuel volatility, spray characteristics, and the burning rate of single drops is treated.

The use of liquid fuels in combustion systems involves the processes of atomization, evaporation, mixing, and burning. In some systems, atomization, evaporation, and mixing of fuel and air may be accomplished sufficiently remote from the flame that the principles of gaseous combustion are applicable. Usually, however, the small available combustion volumes and residence times preclude the separation of these individual steps, and burning occurs in the presence of liquid fuel which acts as a local source of combustible. Although the usual liquid-fuel combustion system involves a simultaneous occurrence of all these steps, much of the published research deals with only the individual steps in the use of liquid fuels. This is necessary because of the great complexity which results when the problems of atomization and evaporation are added to the already difficult problem of combustion. The status of

[6] See paper by Simon and Wagner listed on p. 79.

research pertinent to the combustion of sprays is presented with the intention of indicating fruitful areas for further research.

A number of theories have been proposed to explain the mechanism by which smoke is formed in diffusion flames, but at present no definite and completely consistent theory seems to be available. With the belief that still more experimental work might be helpful in understanding the overall mechanism, several experimental projects were undertaken. Smoke formation was investigated from the standpoint of the effect of pressure, fuel type, external airflow, rate, oxygen enrichment, argon substitution in external air, and fuel temperature. The experimental results were interpreted to indicate a possible step involved in the early stages leading to smoke formation. The understanding gained from these experiments, together with information from the literature, has been used to postulate a possible and relatively complete mechanism of smoke formation.

A comparison of the reaction kinetic processes of several aliphatic hydrocarbons with those of heptane and isooctane is presented in Technical Note 3384. This study included work on the behavior of olefines and exploratory work on the effect of surface-volume ratio on the extent of early stage oxidation of the hydrocarbon.

In aircraft operations there is a need for potent fire-extinguishing agents. In addition to potency, the agents must have properties that make them suitable for use in aircraft environments. For example, the corrosiveness, freezing point, toxicity, and electrical conductivity must be considered. The chemical mechanism chosen for study is that of chain-breaking reactions between agent and active particles (hydrogen and oxygen atoms and hydroxyl radicals). The action of halogenated agents in preventing flame propagation in fuel-air mixtures in laboratory tests is discussed in Technical Note 3565.

Aerodynamic mixing is an important factor in fuel-air preparation, combustion, and exhaust-gas mixing in jet-engine combustors. Because of its importance, it is of interest to investigate aerodynamic mixing under flow conditions commonly found in combustors. Although jet-engine combustors often have extremely intense sound fields, little attention has been given to the effect of sound-wave disturbances on aerodynamic mixing. Technical Note 3760 describes a theoretical and experimental investigation of the aerodynamic mixing of heat by standing sound wave downstream of a continuous line source.

In many technical problems involving heat transfer, the thermal conductivities of gas mixtures are required, often under conditions in combustion systems where measurement is difficult or impossible. An empirical technique for estimating conductivities of mixtures of nonpolar gases is presented.[7] Thermal conductivities of a mixture may be estimated quickly; only the composition of the mixture and conductivities of component gases are required. Accuracy is comparable to that of the more complicated methods which have been previously proposed.

A knowledge of the combustion temperature or of the quantity of fuel required to obtain a specified combustion temperature is necessary in the performance analyses of turbojet and ram-jet engines. With dissociation, the ideal combustion temperature is dependent on combustion-pressure level. Nomographic charts, from which the ideal temperature rise or the ideal quantity of fuel required for a specified combustion temperature may be obtained for a comprehensive range of operating conditions of turbojet- and ram-jet engines, have been prepared. These charts are applicable only to a fuel having a hydrogen-carbon mass ratio of 0.168, which closely approximates the fuel presently used in most turbojet and ram-jet engines. The charts are based on a constant-pressure adiabatic combustion process covering a range of fuel-air ratios from 0 to 1.2 fraction of stoichiometric, a range of combustion pressures from 1/16 to 64 atmospheres, and a range of inlet-air temperatures from 400° to 1600° R.

Combustion-Chamber Research

In the calculation of total-pressure loss and liner airflow distribution for turbojet and can-type ram-jet combustors, it is essential to know the discharge coefficients of combustor-liner wall openings. A study was made to determine the effects of various geometric and flow factors on the discharge coefficients for circular holes having flow parallel to the plane of the hole (Technical Note 3663). The geometric and flow factors considered were hole diameter, wall thickness at the hole, parallel-flow duct height, boundary-layer thickness, parallel-flow velocity, static-pressure level, and pressure ratio across the test hole.

Because the effectiveness of the turbojet engine increases with flight speed and altitude, the service requirements of turbojet engines demand operation at even higher and higher altitudes and flights speeds. The problem of maintaining high combustion efficiency is one of the most important problems of altitude operation. The research approach to the problem has involved both systematic investigations of the effect of individual variables on combustor performance and attempts to relate fundamental combustion parameters such as fuel-spray characteristics, ignition limits, and flame speeds to the observed combustor performance.[8]

While performance studies of various gas-turbine combustors have yielded a large amount of combustor performance data, no method has been presented for

[7] See paper by Brokaw listed on p. 76.

[8] See paper by Olson, Childs, and Jonash listed on p. 78.

correlating these data in terms of fundamental parameters. An empirical correlation of the combustion efficiency data previously obtained in experimental investigations with fourteen aircraft gas-turbine combustors has been developed. A theoretical analysis based on the kinetics of a bimolecular chemical reaction is also presented which, with additional assumptions regarding the mechanism of the combustion process, yields the correlation parameter previously obtained by empirical means.

The low pressure and low temperature encountered at altitude result in reduced combustion efficiency. An explanation of this effect in terms of basic processes is one of the goals of fundamental combustion research. Therefore, an investigation was made to determine the importance of molecular-scale processes in the overall turbojet combustion process. Attempts were made to correlate combustion efficiency with selected fundamental combustion properties and with a simplified reaction-kinetics equation.

In order to determine the relative importance of the basic processes involved in the overall turbojet combustion mechanism, a study was conducted of the effect of oxygen concentration of the inlet oxygen-nitrogen mixture on the combustion efficiency of a J33 single combustor operating with gaseous propane fuel.

Experimental investigations with both turbojet and ramjet combustors have shown that combustion efficiency is adversely affected by the high velocities at which these combustors are required to operate and by the low pressures and low inlet temperatues encountered at high altitudes. A theory of the jet-engine combustion process is therefore needed in order to explain these effects and to indicate the design approaches that are most promising for alleviating these adverse effects. A preliminary theoretical treatment of the combustion process as it occurs in turbojet combustors has been completed.

The trend in afterburner and ramjet-combustor development for increased thrust has resulted in combustors that operate at higher pressures, higher combustion temperatures, and higher velocities. With the development of these high-output combustors, the phenomenon known as screech has been widely encountered. Screech derives its name from the high-pitched audible sound produced. Other manifestations of screech are high-frequency pressure oscillation in the combustor and an increased rate of heat transfer, which have resulted in rapid deterioration or failure of combustor shell, flameholder, and other combustor parts. Screech instrumentation has therefore been developed and the mechanism of screech studied in a 6-inch-diameter simulated afterburner (Technical Note 3567).

Violent lateral oscillations which often occur in the combustion chambers of rocket and jet engines are analyzed.[9]

[9] See paper by Maslen and Moore listed on p. 78.

The formation of carbon in turbojet combustors presents a number of major engine operational problems. Carbon deposition on walls, fuel-injector nozzles, and ignitors affects combustion efficiency, altitude operational limit, and ignition characteristics of the combustor. Also, the distorted airflow and fuel-flow patterns which result from carbon deposits frequently cause warping and burning of the combustor liners. Carbon dispersed in the gas stream as smoke would also be objectionable in military operations where smoke trails remaining in the wake of jet-powered aircraft may be easily detected. An investigation was therefore conducted to determine the effect of systematic variations in inlet-air and fuel parameters on the smoking characteristics of a single tubular turbojet-engine combustor.

Research has been conducted on design and operating factors which may affect ignition limits of combustors and improve the altitude starting characteristics of turbojet engines. Prior studies have evaluated the influence of fuel volatility, spark-plug location, and spark energy on ignition characteristics in both full-scale turbojet engines and single-combustor installations. In addition to these factors, still another variable, spark-repetition rate, must be considered in the study of ignition limits. Data have been obtained for two fuels of different volatility, two spark-energy levels, and three airflow rates in the range of altitude engine-windmilling conditions.

An understanding of the basic processes controlling combustion in a ram-jet engine is one of the ultimate goals of research. Research into the combustion mechanism was continued by determining the variation of combustion efficiency with several pure fuels and fuel blends in a simple ram-jet-type combustor. Combustion-efficiency data were obtained for a 5-inch ram-jet-type combustor employing a simple V-gutter flameholder over a range of inlet static pressures and velocities.

Combustion in a ram-jet engine, or other engines requiring a similar high-speed combustion, may be considered to be a stepwise process. An attempt to evaluate the importance of any one of the steps in controlling the rate of the overall process is obviously difficult in a system in which all the steps vary simultaneously. It is possible to investigate the combustion process separately from the fuel preparation by the use of homogeneous fuel-air mixtures. Such an experimentally idealized system does not necessarily represent a practical ram-jet-type combustor, but will contribute to a better understanding of high-speed combustion processes for such an application. Combustion-efficiency data have been obtained from a 5-inch-diameter combustor employing a straight V-gutter flameholder and a simple cone flameholder. The data obtained cover a range of inlet static pressures, temperatures, and velocities for four fuels.

The maintenance of stable and efficient combustion with low-drag combustors at high heat-release rates is an important objective. The phenomenon of surface combustion and its coincident high reaction rates is well known; however, the process normally involves high pressure losses, which prohibit the conventional use of surface combustion in a ram-jet combustor. The beneficial effects on stability limits and combustion efficiency produced by the application of surfaces immersed in the combustion zone to a ram-jet type burner have been determined. Investigations were made with one, two, three, and four rows of wedges at simulated sea-level and at altitude subsonic ram-jet flight conditions.

A study was made of combustor design factors leading to a sufficiently high heat release and low internal drag to power a high-thrust, low-drag ram-jet engine. Two design principles investigated were the use of flameholders that employed incandescent surfaces heated by immersion in flame and the elimination of the separate fuel-air mixing length by injecting the fuel directly at the upstream end of the flameholder. The scope of this investigation included a series of flameholders designed to provide a high combustion efficiency in a 4- by 8-inch ram-jet combustor 24 inches long at an inlet-air velocity of 200 feet per second, inlet-air pressure of 60 inches of mercury absolute, inlet-air temperature of 200° F, and near stoichiometric fuel-air ratio.

Several investigations have been made on the operational characteristics of the V-gutter flameholder. These studies demonstrate that fuel distribution exerts an important influence on combustor performance. At lean overall fuel-air ratios, the fuel distribution has a greater effect than that of flameholder geometry upon combustor performance. The primary objective of this work was therefore to determine the effect of combustion on the diffusion coefficient in the fuel preparation zone and to design, with the aid of this information, a combustor that will provide the fuel-air distribution necessary for efficient combustion at lean fuel-air ratios.

Jet-engine combustion-chamber design must conform to many requirements specified by carbon deposition, liner temperature, size, combustion stability, temperature profile, pressure drop, and combustion efficiency. Many of these requirements may be evaluated by inspection and simple measurements, but evaluation of temperature profile, pressure drop, and combustion efficiency entails more elaborate research instrumentation. An automatic polar-coordinate traversing system for determining jet-engine combustor performance was devised (Technical Note 3566). A combined temperature and pressure probe is swept circumferentially through a quarter-annular exhaust duct at selected radial positions. Data are recorded as a function of probe position. This method furnishes complete temperature, pressure, and flow profiles with a single probe.

LUBRICATION AND WEAR

Fundamentals of Friction and Wear

The lubricated parts of turbine engines and aircraft components are required to operate at continually increasing temperature levels. This trend is unavoidable where higher performance engines and greater flight speeds are the ultimate objectives. Adequate bearings, seals and lubricants must be available for operation at these high temperature levels.

To fill the lubricant need, silicone-diester blends were formulated on the basis of fundamental studies on lubrication with silicones. Because it had excellent viscometric properties and relatively good thermal stability, one blend (SD–17) was considered as a possible turboprop-engine lubricant. It was found to be a very good gear lubricant. The blend performed satisfactorily in more than 17 hours of operation at moderate power levels during a functional check in a T–38 turboprop engine. However, subsequent studies showed the blend to be subject to foaming at bulk oil temperatures (300° F) necessary for high-performance turbojet engines. Under foaming conditions and at high temperatures, excessive oxidative and thermal degradation occurred.

Halogen-substituted methane and ethane gases continue to be of interest as high-temperature boundary lubricants. Because the functioning of these gases depends on chemical reaction between chlorine atoms of the decomposed gas and the surfaces being lubricated, the bearing materials are very important.[10] With steel surfaces, the best results were obtained when both slider surfaces were of nearly the same hardness. Modified H-Monel and beryllium copper were successfully lubricated by difluoro-dichloromethane; in both cases the mating material was hardened tool steel. Other common bearing materials such as silver were not lubricated by the gas. Boundary lubrication by gases appears feasible for temperatures up to 1,000° F.; few known liquids have any promise for lubrication of bearing surfaces operating at temperatures above 700° F.

Because solid lubricants are among the most promising high-temperature lubricants, an experimental friction study was made using graphite and mixtures of graphite with lead oxide, cadmium oxide, sodium sulfate, or cadmium sulfate as solid lubricants. Runs were made at temperatures to 1,000° F. with various steel and Inconel combinations (Technical Note 3657).

Graphite powder lubricated metal surfaces at temperatures sufficiently high to promote oxidation of the

[10] See paper by Murray, Johnson, and Swikert listed on p. 78.

metal surfaces. At intermediate temperatures, graphite powder alone failed to function as a lubricant. Graphite would not lubricate cast Inconel on Inconel X at temperatures between approximately 150° and 800° F. Similarly, with steel, which oxidized more readily than Inconel, the temperature below which graphite would not lubricate was 475° F. as compared with 800° F. for Inconel. Mixtures of some metallic compounds and graphite were effective lubricants from room temperature to 1,000° F. In some room-temperature experiments, CdI_2 and $CdCl_2$ were effective solid lubricants for titanium.[11]

The usefulness of liquid lubricants has been seriously impaired by oxidation at engine operating temperatures. One method of minimizing this problem is to obtain more effective sealing of the lubrication system and thereby limit the amount of air to which the lubricant is exposed. Sliding-contact seals are necessary for closed lubrication systems. Wear and friction studies were therefore made to show the effects on performance of temperature, type of mating material, and minor composition changes in typical carbon-seal materials (Technical Note 3595). Wear of carbon-seal materials increased rapidly with temperatures between 400° and 700° F. The effect of temperature on wear was reduced by using chromium-plated steel as the mating surface rather than stainless or tool steel. In general, the type of carbon and impregnation of the carbon-seal material had little effect on wear compared with the effect of the mating material.

Bearing Research

The effect of air and nitrogen atmospheres on the temperature limitations of liquid and solid lubricants was studied using small ball bearings.[12] All lubricants tested in both nitrogen and air atmospheres were effective to higher temperatures in a nitrogen atmosphere, and the solid lubricants (molybdenum disulfide and graphite) were best. Both graphite in air and molybdenum disulfide in nitrogen were effective to 1,000° F. These studies, which showed that the limiting temperatures of liquid lubricants could be extended by preventing oxygen from the atmosphere from contacting the lubricant, led to further tests to determine the role of oxygen. The results of an investigation to determine the effect of oxygen concentration in the atmosphere on oil lubrication of small ball bearings at high temperatures are reported.[13] The results showed that a 20-millimeter-bore ball bearing will run at temperatures to 850° F. with oil lubrication if the oil flow exceeds a critical amount.

Early detection of bearing failures in bearing research is important because detection of incipient failure early enough can often avoid a total failure, and additional information on the cause of failure can be obtained. In contrast to the slow response of temperature, the second derivative of temperature with respect to time (the acceleration) reaches a high positive value before significant temperature changes are noticed. Accordingly, an instrument for measuring the magnitude and sign of temperature acceleration was developed and found to be effective in detecting incipient bearing failures in 75-millimeter-bore roller bearings.[14] Incipient failures were reproduced by shutting off the oil flow, and in each test it was found that temperature acceleration gave a faster warning than temperature, motor torque, or noise level. Total failures were thus avoided when using temperature acceleration as a warning device.

An experimental investigation of eccentricity ratio, friction and oil flow of long and short journal bearings was conducted at Cornell University under NACA sponsorship and is reported in Technical Note 3491. The data provide charts of plain bearing performance which cover the range of length-diameter ratio of ¼ to 2.

COMPRESSORS AND TURBINES

Compressor Research

As part of the investigation of the fundamental nature of compressor stall, a theory of stall propagation was developed and reported in Technical Note 3580. Experimental results obtained from a stationary circular cascade and a single-stage axial-flow compressor indicate that the theory predicts propagation velocities within 25 percent over a wide range of wave lengths.

To obtain a better understanding of the stall phenomenon by providing detailed information regarding the stall mechanism, pressure, temperature, and flow fluctuations, an investigation of the rotating-stall characteristics of a compressor with a 0.9 hub-tip radius ratio was undertaken. A study of a rotor having a 0.9 hub-tip ratio (no guide vanes or stators) showed stall patterns consisting of two, three, and one total-span stall zones developing in that order upon reduction of flow coefficient. The one-stall-zone pattern caused the most severe pressure, temperature, and flow fluctuations (Technical Note 3518). The rotating-stall characteristics of the 0.9 hub-tip-ratio stage were extended to include several guide-vane rotors and guide-vane rotor-stator configurations. Guide vanes having turning angles of −22.5°, 0°, 22.5°, and 40° were used. Results indicate that the initial rotating-stall point, number of stall zones formed, and stall propagation rate of a configuration depend not only upon blade-row inlet angle and pressure-rise characteristic but also upon the overall characteristics of the multiblade row unit as well. The addition of stators to a particular guide-vane-rotor configuration generally increased the flow coefficient where rotating stall was initially encountered

[11] See paper by Peterson and Johnson listed on p. 78.
[12] See paper by Nemeth and Anderson listed on p. 78.
[13] See paper by Nemeth and Anderson listed on p. 78.
[14] See paper by Schmidt and Anderson listed on p. 78.

and lowered the stall propagation rate. The number of stall zones formed in the annulus did not appear to depend upon guide-vane turning or the presence of stators (Technical Note 3711).

Incompressible two-dimensional flow was analyzed specifically to establish simplified equations and considerations that may prove useful in the estimation of profile losses and in the correlation of experimental data in the low-speed two-dimensional cascades. The relative importance of the various factors entering the loss relations was evaluated and relations were obtained for the mixing-loss ratio and for the effect of trailing-edge thickness (Technical Note 3662).

Information was obtained (Technical Note 3568) on the average pressure indicated by a total-pressure probe subjected to a stagnation pressure that alternates periodically between two constant values. The errors were reduced when the probe design was such as to ensure laminar-flow pulsations in the probe at all times. The averaging error was minimized when the inside diameter of the probe entrance tube was made as small as possible and its length as great as possible consistent with an acceptable time lag.

Turbine Aerodynamics

The smoke-visualization studies of the secondary-flow in turbine-blade passages have been continued by investigating secondary flow in the rotor-blade tip region of a low-speed turbine. The present investigation (Technical Note 3519) was conducted to obtain a visualization of the tip-flow phenomena and to determine which factors of blade geometry and tip condition influence the types of secondary-flow behavior encountered at the blade tip. Results of the investigation include qualitative information on tip-clearance flow, crossflow, and scraping flow and should aid in extending the study to higher air speeds.

An analysis was derived (Technical Note 3651) for three-dimensional boundary-layer flow over a flat surface with a leading edge under main-flow streamlines which are representable by polynomial expressions. The boundary layers are laminar and incompressible, and the profiles of their velocity components have similarity with respect to their rectangular coordinates. Solutions were obtained for main-flow streamlines representable by polynomials up to the eleventh order. Flow-visualization experimental checks of the theory are provided for several flow configurations and for comparisons of the behavior of thick and thin boundary layers.

One of the principal objectives of turbine research is a better understanding of the fundamental nature of the flow and the loss sources encountered in turbomachine blade rows. The effect of compressibility on the loss characteristics downstream of two-dimensional turbomachine blade rows is therefore analyzed in Technical Note 3515. Equations are derived for obtaining the compressible-flow boundary-layer characteristics for a simple power-velocity distribution. Loss coefficients at the blade trailing edge are then obtained in terms of these characteristics. Finally, overall loss coefficients, including the effect of mixing downstream of the blade row, are obtained in terms of these characteristics.

Turbine Cooling

Gas-to-gas heat exchangers find many uses in modern aircraft and missiles. Usually, a large number of calculations must be carried out for the design of such a heat exchanger. It is, therefore, of advantage to have a simple method available by which the dimensions of the heat-exchanger core can be rapidly calculated. Such a calculation procedure is described in Technical Note 3655. The dimensions of the core of a gas-to-gas crossflow heat exchanger with prescribed heat-transfer surface can be determined rapidly.

The problem of selecting heat-exchanger configurations for optimum performance was investigated. The fluid on one side of the exchanger was assumed to have negligible heat-transfer resistance, and the amount of heat exchanged per unit time and the mass flow and inlet state of each fluid were prescribed. Any one of the parameters, power expended, weight, volume, or frontal area, can be optimized with respect to any one of the three remaining parameters when the heat exchanger is arranged normal to the approaching primary fluid. When the heat exchanger is inclined at an angle to the upstream direction, any one of the parameters, power, weight, or volume, can be optimized with respect to any one of the two remaining parameters. With this arrangement, the projected frontal area of the inclined heat exchanger will be equal to that of the heat exchanger requiring the minimum duct cross-sectional area when arranged normal to the primary fluid flow (Technical Note 3713).

Experimental investigations of free-convection effects on heat transfer for fluids flowing vertically through tubes with small and large length-to-diameter ratios were made by NACA and the Massachusetts Institute of Technology, respectively. However, the data from the two investigations were not analyzed on the same basis. In Technical Note 3584, therefore, experimental heat-transfer data for the turbulent flow of fluids through stationary vertical tubes with both small and large length-to-diameter ratios (to 40) are compared. The limits of the different regions, originally established for a tube with small length-to-diameter ratio, apply on the basis of existing data to a tube with large length-to-diameter ratio.

The literature contains many investigations on wedge-type flow, including results for fluids with different Prandtl numbers flowing through porous flat plates. A summary of exact solutions of the laminar boundary-layer equations for wedge-type flow, useful in estimating heat transfer to such arbitrarily shaped bodies as turbine blades, is presented in Technical Note 3588. The principal results are restricted to a Prandtl number of 0.7 at the wall and are presented for large temperature changes through the boundary layer adjacent to a constant-temperature wall and for small temperature changes through the boundary layer adjacent to a wall of either a constant or a variable temperature.

Blade Vibration and Flutter

The problem of the failure of compressor blades in the stall region has become more acute in recent years with the greater use of high-compression-ratio engines. A study of the vibrations occurring in the stall region is described in Technical Note 3581. A compressor-type annular cascade of airfoils was used to determine the effects of varying the angle of attack, blade chord and spacing, number of blades, and air velocity on the vibrations.

The phenomenon of rotating stall or stall flutter in axial compressors is discussed in Technical Note 3571 by means of an unsteady boundary-layer analysis.

Heat Transfer

Knowledge of the flow characteristics of various liquids in passages having both circular and noncircular cross sections is important in the removal of heat from nuclear reactors. An experimental investigation of the flow patterns in a narrow vertical enclosure was therefore made. A shadowgraph was used in the investigation to study the free-convection flow of water in the narrow enclosure, which was cooled at the top through a copper surface and open at the bottom to a heated reservoir. This visual inspection of flow patterns yielded information on the steadiness of flow patterns with time, the size and uniformity of the various upflow and downflow regions, and the direction of flow velocities with particular attention to nonvertical components. The dominating characteristics of the flow patterns were instability and change.

It is often desirable to use short passages in heat exchangers in order to take advantage of the high heat-transfer coefficients in the entrance region. The effect of various factors on the turbulent heat transfer and friction in the entrance regions of smooth passages was investigated analytically.[15] The influence of Reynolds number, Prandtl number, initial velocity distribution, wall boundary condition, passage shape, and of variable fluid properties was predicted. The results indicated that approximately fully developed heat transfer and friction are, in general, attained in an entrance length of less than 10 diameters. Substantial agreement between analysis and experiment was obtained for heat transfer to air in the entrance regions of tubes and parallel plates.

[15] See paper by Deissler listed on p. 77.

ENGINE PERFORMANCE AND OPERATION

Turbine Engines for Helicopters

The gas-turbine engine shows considerable promise as a solution to the propulsion requirements of the high-performance helicopter. These requirements are (1) high ratio of rotor power available to engine weight, (2) speed-power characteristics that permit a relatively free choice of rotor tip speeds, (3) satisfactory fuel economy over a wide range of speed and power settings, and (4) dynamic response characteristics that are suitable to transitional flight operations. In a number of prior investigations these requirements were shown to be more nearly satisfied by the free-turbine engine than by the fixed-turbine engine.

An analysis of fixed- and free-turbine engines applicable to helicopter propulsion was made. Calculated performance characteristics of the two engines are presented in Technical Note 3654 in terms of the appropriate equivalent parameters. Performance comparisons are drawn for off-design-point and altitude conditions of engine operation. The behavior of the engines during rapid power and speed modulation and their dynamic response characteristics when coupled to a helicopter rotor are also examined.

The flow with heat addition in which a body force acts transverse to the flow is treated in Technical Note 3594. The results may be applied to flow with heat addition in pipe bends, through helicopter rotor-tip combustors, and through ram jets in abrupt maneuvers.

Foreign-Object Damage

Severe damage to jet engines may result from the impact of foreign objects on compressor and turbine blades. An investigation was therefore undertaken to determine the effect of typical impact damage on the fatigue strength of jet-engine compressor blades. First-stage compressor-rotor blades from a production engine which had suffered foreign-object damage were fatigue tested at the endurance limit of the blade material. The number of cycles to failure was correlated with the depth, location, frequency, and type of damage. The most serious damage to the blades, as measured by the reduction in fatigue strength, resulted from nicks at the leading and trailing edges in the vicinity of the maximum-vibratory-stress section of the airfoil. The strength of dented blades could be restored by reworking, but seriously nicked blades could not be reliably restored (Technical Note 3275).

Ground and Flight Investigations of Ram-Jet Engines

An investigation was conducted in a blowdown jet of a 6.5-inch-diameter ram-jet engine at Mach numbers of 1.81 and 2.00. The engine was demonstrated to have wide combustion limits, reliable ignition characteristics, and good combustion efficiency when operated on ethylene (C_2H_4) fuel. The same type of engine was used to perform flight tests. The flight investigation covered Mach numbers from 1.9 to 3.0 and altitudes from 1,800 to 40,900 feet. Good agreement with the static test performance was obtained. The engines operated reliably until the fuel was expended. During the flight tests, the engines accelerated the test vehicle at a maximum acceleration of 3.6*g*.

Miniature Ram-Jet Engine

In order to facilitate the determination of jet effects in wind-tunnel model tests at supersonic speed, a small (1.1-inch-diameter) ram-jet was designed and its performance was investigated. The engine, which burned a gaseous fuel, was operated over at Mach numbers from 1.42 to 2.28 and at Reynolds numbers which were equivalent to an 18-inch-diameter ram-jet flying at an altitude of 74,000 feet. Reliable operation was obtained over a wide range of thrust coefficients and fuel-air ratios. Spark ignition was demonstrated to be rapid and reliable.

Thrust Reversal

Many uses for thrust reversers on jet aircraft have been proposed. They include braking the landing roll, reversing or spoiling thrust during the landing approach so that maximum engine speed may be maintained, and braking during diving maneuvers to limit flight speed. To be used effectively, the reverser must give the desired amount of reverse thrust without affecting engine operation. Also, the design must lend itself to stowage with a minimum amount of boattail or base drag.

As part of an overall investigation of thrust reversers and their associated problems, studies were conducted with cold flow on thrust reverser models. This work was done on a small-scale unheated-air-duct setup equipped with a 4-inch-diameter exhaust nozzle.

The performance of a hemispherical thrust reverser over a range of geometric variables and some of the factors that affect reverse-thrust performance have been obtained. The effects of several simplifications to the hemispherical design are also shown. The effects of most of the design variables of the reverser were obtained at an exhaust-nozzle total- to ambient-pressure ratio of 2.0. The basic data over a wide range of conditions are also included, however, so that other comparisons may be made.

The performance of cylindrical-type thrust reversers and the effects of several modifications on their performance were investigated at an exhaust-nozzle pressure ratio of 2.0. These modifications include changes in frontal area, width-to-height ratio, depth, lip angle, end-plate depth, and end-plate shape. The performance of swept-type cylindrical thrust reversers, the relation of reverse-thrust ratio to reversed-flow attachment, and thrust-modulation characteristics were also investigated.

Preliminary data were obtained on the performance of several cascade-type thrust reversers located upstream of the exhaust nozzle up to an exhaust-nozzle pressure ratio of 2.4. Such reversers are herein referred to as the tail-pipe-cascade type. A total of 15 different tail-pipe-cascade configurations were investigated. These included two blade shapes, several cascade length-to-span ratios, and various innerbody lengths. Basic airflow characteristics and reverse-thrust ratio are plotted against exhaust-nozzle pressure ratio for all 15 configurations at full reversal. For some of the configurations, modulation performance and surveys of total pressure and flow angle were obtained at the cascade discharge.

Technical Note 3664 presents the types of thrust reversers investigated under the overall NACA program, summarizes the important performance characteristics, and presents proposed operation methods. Three types of reversers were investigated, target, tail-pipe cascade, and ring cascade. The effects of design variables on performance, reversed-flow fields, and thrust-modulation characteristics were determined for each type.

The performance of a hemispherical target, which is a basic thrust-reverser type, was evaluated in full-scale tests (Technical Note 3665). A turbojet engine equipped with such a device was pylon-mounted under the wing of a cargo airplane, the installation simulating that on a jet bomber or transport. The thrust reverser was operated at both stationary and taxi conditions, but the airplane was not flown. In addition to obtaining the performance of the thrust reverser, the heat-rise patterns and rates resulting from impingement of the reversed hot gases on a simulated lower wing surface were also measured. Because, during stationary operation, some of the hot reversed gases penetrated as far forward as the engine inlet and were reingested, taxi tests were conducted to estimate the ground speeds required to disperse the reversed gas flow and prevent reentry into the engine inlet.

Ducts

Size and weight penalties of air-induction systems would be decreased by the development of efficient short subsonic diffusers. Preliminary tests on a series of such diffusers employing various flow control devices have been completed.

Engine Controls

In order to facilitate consideration of control systems for two-spool turbojet engines, a brief analysis of the linear response characteristics of this type of engine is presented in Technical Note 3274. The analysis is concerned with the linear responses of the two spools to changes in turbine-inlet temperature at constant exhaust-nozzle area and to changes in exhaust-nozzle area at constant turbine-inlet temperature.

General equations of response are developed from linearization of functional relations. The general equations are then evaluated at design speed by means of representative engine thermodynamic relations. The resultant equations are corroborated with experimental data.

POWER-PLANT MATERIALS

High-Temperature Materials

Improved performance of turbojet engines and the realization of nuclear-powered weapons is to a large degree dependent on the development of materials capable of withstanding severe conditions of corrosion, temperature, and stress. Research is continuing in an effort to improve existing materials and to develop new materials to fulfill these conditions.

Many intermetallic compounds have for some time been of interest for high-temperature application. However, there is very little generalized knowledge by which the potentialities of materials in this category can be evaluated. To build up a background of information from which theories can be formulated, as well as to provide specific data on an intermetallic of interest, Ni_3Al was studied. This material is of particular interest because of its high melting point and stability at high temperatures. An investigation (Technical Note 3660) of the effects of homogenization and of composition on the tensile properties of as-cast Ni_3Al intermetallic phase alloys showed that the tensile strengths of these alloys at both room and elevated temperature are very sensitive to composition, structure, and grain size.

For high-temperature applications, many brittle alloys, cermets, and ceramics are of interest. In order to utilize these materials in advanced turbojet engines, improved methods of fastening, designed to reduce bending stresses and to minimize stress concentrations, are required. An improved cermet bladed design, utilizing a curved root, was developed. This design eliminates the need for a blade platform and results in significantly reduced stresses.

Recent efforts to develop alloys of superior elevated-temperature creep resistance have heightened interest in the function of grain boundaries in creep, both because coarse grained materials are found to exhibit greater creep resistance at high temperature and because high-temperature creep is usually intergranular. The gliding of one metal crystal with respect to another parallel to their mutual grain boundary has been studied in pure aluminum bicrystals (99.95% Al, balance copper, iron, chromium, silicon, and magnesium) during isothermal creep at temperatures ranging from 200° to 650° C under static stresses from 10 to 1,600 psi (Technical Note 3556). The mechanism of grain boundary gliding was found to be a coordinated alternation of slip and recovery in a chain of subgrains along the grain boundary which was highly sensitive to crystal orientation. It was postulated that the addition of alloying elements should affect the process most markedly in those respects relating to the occurrence of recovery. Consequently, studies of the effects of Cu (0.1 to 3 percent) on Al were made (Technical Note 3678). It was found that the minimums in stress and temperature, below which grain boundary motion does not occur, increase regularly with the copper content, as would be expected if recovery is necessary for movement. Otherwise, the effects, if any, of the copper solute upon grain boundary displacement and its rate were too small for identification by the experimental technique employed.

The mechanism responsible for the bonding of ceramics to metals has been under study for several years. Earlier work was concerned with the manner in which adherence develops when cobalt ions are present in the coating, using radioactive cobalt 60 as a tracer technique. A similar study of the effect of nickel dipping on adherence has been conducted using radioactive nickel 63, produced by pile irradiation of highly purified cobalt-free nickel (Technical Note 3577). It may be concluded that the nickel from the nickel dip remains as metal at the enamel-metal interface as it does not oxidize during the firing treatment. Further, the presence of nickel from the nickel dip had little or no effect on the deposition of cobalt metal during firing. A further study of the influence of copper ions on adherence of vitreous coatings to stainless steel has been completed (Technical Note 3679). In general, the presence of copper ions in the coating produced a significant increase in adherence. However, this effect decreased with increased firing time and temperature.

Stresses Research

The thermal-shock problem is of vital importance in jet engines, nuclear powerplants, rockets, and high-speed missiles subjected to aerodynamic heating. Since structural damage may be caused by rapid temperature changes, there is a need for a better understanding of the factors affecting the thermal-shock resistance of materials. Two theories predicting the thermal-shock resistance of brittle materials are described and compared with experimental results.[16] Equipment for testing specimens over a wide range of heat-transfer coefficient is described and four independent methods of determining the thermal-shock parameter are shown. A simple approximate method for computing transient thermal stresses in hollow cylinders, plates, and hollow spheres is reported.[17] These bodies were used to approximate the more complex configurations found in practice. The method

[16] See paper by Manson and Smith listed on p. 78.

[17] See paper by Mendelson and Manson listed on p. 78.

makes use of polynomial approximations to the temperature distribution. Using these approximations reduces the partial differential equations of the problem to first-order ordinary differential equations, making possible practical solution of the problems in relatively little time.

In tension or creep testing, misalinement between the specimen axis and the loading axis can influence the results by causing a bending stress to be superimposed on the applied tensile stress. This bending stress is particularly important when materials of limited ductility such as some of the very high-strength creep-resisting alloys and cermets are used. Conventional testing equipment permits considerable misalinement, which varies in an uncontrollable manner from test to test. This variation introduces scatter into the test results. A new axial-loading creep machine was developed that reduces the misalinement to a minimum and represents a large improvement over conventional equipment.[18]

Physics of Solids

In order to continue the advanced development of materials for aircraft applications, a basic knowledge of the solid state must be acquired. Physics-of-solids research is designed to gain a greater understanding of solids on an atomic and microstructural level in those areas related to the mechanical, corrosion, and temperature properties of materials.

Because of the presence of oxygen atoms in rocket-engine and reactor atmospheres, it is important to consider the effect of oxygen atoms on the oxidation rate of metals. An investigation has been made of this effect on platinum. The rate of surface oxidation was found to be markedly increased by the presence of oxygen atoms. Surface oxidation was examined at 1000° C and a pressure of 0.50 millimeter of mercury under such conditions that ionic sputtering was insignificant. The reaction was found to obey a linear law, and oxygen atoms were shown to be at least 400 times more reactive than oxygen molecules.[19] In studies of the oxidation of metals at high temperatures, an accurate method of measuring the extent of oxidation is necessary. The conductometric method has been successfully applied to the oxidation of iron at approximately 600° C. Iron ribbons of known thickness were used in the tests and the conductance of the central portion of the specimens was measured by the potentiometer method. A constant current of 50 milliamperes, small enough to prevent heating even in vacuum, was used. The conductometric method gave a true measure of the extent of oxidation, which is in agreement with the gravimetric method. An analytical method of measuring the amount of unoxidized iron remaining in the specimens is briefly described.

[18] See paper by Jones and Brown listed on p. 77.
[19] See papers by Fryburg listed on p. 77.

It has been known for many years that surface cracks play a large part in decreasing the strength of materials below the theoretical maximum value. The presence of such cracks was previously inferred by indirect methods. It is shown that the crystallization behavior of metallic films deposited on alkali-halide, single-crystal surfaces allows a direct method of observing the formation of cracks on surfaces.[20] This method was used to measure the rate of crack formation at various temperatures.

The nature of solid surfaces is important in determining the physical properties, such as strength, of solids. An investigation was conducted to clarify the effects of radiation on the surface of brittle materials. Electron-diffraction studies have shown that after sufficient irradiation the surfaces of sodium chloride crystals break up into small crystallites, which exhibit preferred orientations. The irradiation time required to produce surface damage is much greater for water-polished crystals than for untreated ones. The results are discussed on the hypothesis that the large increase in vacancy concentration accompanying F-center formation may aid in relieving strains and result in reorientation of blocks of ions.[21]

The effect of vacancy and F-center concentration, as produced by X-irradiation, on the room-temperature creep properties of NaCl single crystals has been studied. Under a load of 1.200 grams per square millimeter, logarithmic creep behavior was noted for both annealed and quenched crystals before or after irradiation. Irradiation with 50-kilovolt X-rays was found to have a marked effect on both initial deformation and subsequent creep, causing an initial softening followed by hardening as irradiation is continued. The behavior after irradiation can be explained on the basis of the changes in vacancy concentration and distribution that accompany F-center formation. Experiments in which the vacancy concentration was changed by quenching from various temperatures indicate that the creep rate may depend in a simple way on the number of vacancies.[22]

Further work on color-center precursors is reported. It was previously found that NaCl crystals which had undergone electrolysis, but which remained colorless, were much more sensitive to irradiation with X-rays. Recent work shows that these same properties exist in NaCl crystals when they are colored additively and then bleached by electrolysis.

An insight into the nature of the state of cold-worked alloys is provided. Measurements were made of the resistivity and thermoelectric power of samples of AuCu in various nonequilibrium states. One set of samples was disordered by quenching from 750° C, and annealing curves were obtained at various tempera-

[20] See paper by Metz and Lad listed on p. 78.
[21] See paper by Leider listed on p. 78.
[22] See paper by Lad and Metz listed on p. 77.

tures. A second set was first ordered and then cold-worked essentially to complete disorder, and annealing curves were taken at several temperatures. It follows from the detailed results that to specify the state of an alloy, it is necessary to give, in addition to resistivity and thermoelectric power, at least one other quantity such as the coefficient of magnetoresistivity.

Near the melting point, solids may be disordered sufficiently so that they could be described by the methods-of-liquid theory. One of the difficulties in liquid theory is the computation of the potential energy of a molecule in the liquid. A method of computing this energy has been devised which is based on a knowledge of the radial-distribution function and the intermolecular-potential-energy function. Calculations for argon indicate that the method gives satisfactory results. The unique properties of the MgCd alloy system afford a convenient "testing ground" for the various theories of ordering in binary alloys. An experimental investigation of the specific heat of this system as a function of temperature was carried out with an adiabatic vacuum calorimeter.

In studying the fundamental properties of some semiconducting materials, it was found that the electromotive force developed parallel to the gradient of light absorption in germanium crystal is reduced by the application of a transverse magnetic field.

The logarithmic expression for the temperature dependence of viscosity satisfactorily describes the behavior of a large number of normal liquids but fails when applied to associated liquids. A method is given for deriving the temperature dependence of viscosity by considering the molecule to consist of two force centers, an ordinary van der Waals force center, and a dipole center. The resulting equation is successfully applied to a number of associated liquids. The relation between this equation and vapor pressure is also deduced, and results are given for water.

In designing nuclear reactors, calculation of neutron flux distributions and the critical mass is essential. The fuel elements consist of thin strips of uranium adjacent to moderator material or of solutions of enriched uranium salt in a liquid moderator. The analysis is usually simplified by assuming the fuel and moderator to be homogeneously mixed. In practical assemblies, the core may consist of repetitive "cells" in a fuel-moderator assembly. These are much less than the order of a mean free path, so that solutions of a higher order than diffusion theory are required to evaluate departures from homogeneous conditions. The steady-state diffusion-theory solutions and the next higher order approximation of several self-shielding problems have been obtained for multiregion cells of rectangular and cylindrical geometry (Technical Note 3661). The neutrons were assumed to be monoenergetic, and their distribution function was assumed to be dependent only upon one spatial coordinate. A spherically symmetrical scattering in the center of the mass system was also assumed. The solutions of the diffusion theory and the transport-theory flux equations were obtained by a differential analyzer. Additional results were obtained by approximating the effects of molecular binding in the water molecule. The thermal neutron flux distributions were used to compute the ratio of total absorption in uranium to the total absorption in the cell.

Additional studies of the multiple scattering of slow (less than 100 kiloelectronvolts) alpha-particles were made. These studies had previously been neglected because of the difficulty of introducing the particles into apparatus with known energies. This difficulty was overcome by first passing the alpha-particles through a velocity selector and then through a thin nylon window into the cloud chamber. For charged particles of medium energies, the experimental data and the theory are in reasonable agreement. For low-energy nuclear particles, however, there is no adequate theory and there are very few data.[23]

ROCKET ENGINES

Propellants

A continuing interest in hydrocarbon fuels and liquid oxygen as rocket propellants is assured by favorable logistics and relatively high specific impulse. Theoretical rocket performance for frozen composition during expansion was calculated for the propellant combination of JP–4 fuel and liquid oxygen at two chamber pressures and several pressure ratios and oxidant-fuel ratios.

A knowledge of flame propagation limits is necessary for designing combustors and specifying pressures, compositions, and temperatures in which a given gaseous fuel-oxidant combination will burn. With respect to theory, flame propagation limits are important because they can be correlated with other combustion parameters and thus aid in the fundamental understanding of combustion. Flame-propagation limits of propane and *n*-propane in oxides of nitrogen were obtained (Technical Note 3520) at subatmospheric pressures in a 2-inch-diameter by 48-inch-length tube.

Experiments in which the rocket propellant, crude N-ethylaniline (monoethylaniline) and mixed acid (nitric plus sulfuric), failed to ignite satisfactorily at low temperature indicated the necessity of a knowledge of the self-ignition properties of certain rocket propellants at low temperatures as well as at moderate temperatures inasmuch as rockets may be required to start at high altitudes or under arctic conditions. An investigation was therefore conducted to determine possible rocket fuels that ignite spontaneously at low temperatures with mixed acid (nitric plus sulfuric) in a more reliable manner than crude N-ethylaniline. Experiments were also conducted at sea level and at a pressure altitude of approximately 55,000 feet at various temperatures in

[23] See paper by Allen, Webeler, and Barile listed on p. 76.

order to determine the starting characteristics of a commercial 220-pound-thrust rocket engine using crude monoethylaniline and other fuels with mixed acid.

The freezing points and low-temperature fuel-igniting properties of fuming nitric acids are of current interest because of a demand to extend the use of these oxidants to rockets operating at low temperature. The interrelated effects of water, from 0 to 10 percent by weight, and nitrogen tetroxide, from 0 to 25 percent by weight, in fuming nitric acid were studied with respect to the freezing points of the acid and the ignition delays with several fuels. Several possible chemical causes for the opposing effects of water and nitrogen tetroxide on ignition have been proposed.

Ignition delays of several propellant combinations obtained with a modified open-cup apparatus and a small-scale rocket engine of approximately 50 pounds thrust were compared to study any correlations that might exist between the two methods of ignition-delay determination. The results were used in determining the relative utility of each apparatus.

The literature pertaining to the preparation, physical properties, corrosiveness, thermal stability, constitution, and analysis of various nitric acids has been reviewed primarily with respect to their use as rocket oxidants. Conflicting data are evaluated and recommendations for additional experimental work are indicated.

Numerous studies have been made of the vapor pressure of essentially pure nitric acid and of the binary system, nitric acid-water. Data for the ternary system, nitric acid-water-nitrogen dioxide, are for the most part lacking. Work was therefore undertaken to provide more complete vapor-pressure data for the ternary system at physical equilibrium. Mixtures containing 71 to 97 weight-percent nitric acid, 0 to 20 percent nitrogen dioxide, and 0 to 15 percent water were used.[24]

Because the storage of fuming nitric acids presents a serious operating problem, means for improving the storage properties of this acid were sought. The storage properties of fuming nitric acids, with and without additives, were studied at a temperature of 170° F in closed containers of approximately 100-milliliter capacity; the containers had aluminum bodies and stainless-steel caps.

Among the storage properties of fuming nitric acid, corrosion and decomposition are of foremost concern. Additional information concerning the effectiveness of fluorides as corrosion inhibitors in fuming nitric acid was therefore obtained. It was found that for acids containing no fluorides, the weight loss of aluminum was approximately one-fifth that of stainless steel. Addition of 1-percent fluoride ion to the acid reduced the weight loss of both metals to practically zero even after 26 days of exposure to the acid at 170° F. Additional information concerning the effect of fluorides on corrosion was obtained by measuring the electrode potentials of the metals against a platinum reference electrode.

[24] See paper by McKeown and Belles listed on p. 78.

Rocket Combustion

Ignition-delay determinations of several fuels with nitric-acid oxidants were made at simulated altitude conditions from sea level to 100,000 feet utilizing a small-scale rocket engine of approximately 50 pounds thrust. Included in the fuels were aniline, hydrazine hydrate, furfuryl alcohol, furfuryl mercaptan, turpentine, and mixtures of triethylamine with mixed xylidines and diallylaniline. Red-fuming, white-fuming, and anhydrous nitric acids were used with and without additives.

The rocket phenomenon known as screaming often causes chamber, injector, or nozzle burnout failures and has been observed to increase the specific impulse. Rocket-engine screaming is a type of combustion-driven oscillation, with frequencies from 1,000 to 10,000 cycles per second, and is characterized by an audible wailing exhaust sound, by a bluish almost-invisible exhaust jet in which the shock positions oscillate (making the shock pattern appear fuzzy to the eye) and by increased heat transfer to the chamber surfaces. The high-frequency oscillations have been attributed to a combustion-reinforced pressure wave passing through the chamber and reflecting from the chamber surfaces to trigger the succeeding combustion surge. The frequency would therefore be governed by the velocity of wave propagation and the geometry of the chamber. A simplified analysis, based on the concept of acoustical resonance, has been developed to correlate scream frequencies with chamber geometry in terms of experimentally measurable quantities. The derived parameter is substantially independent of propellant combination or operating conditions.

The application of radiation-measurement techniques to the determination of gas temperatures in the flame resulting from liquid propellant reactions has recently been investigated. Such techniques are desirable in rocket combustion and injector design studies because they permit the study of conditions in a flame zone without disturbing the flow and without the necessity of maintaining a probe in the chamber. Radiation-temperature measurements were made throughout the flame developed within an open-tube combustor using liquid oxygen and a heptane-turpentine mixture as the reactants.[25] The temperature measurement utilizes carbon radiation from the flame.

[25] See paper by Auble and Heldmann listed on p. 76.

AIRCRAFT CONSTRUCTION

Problems associated with the structural integrity of aircraft in the subsonic and lower supersonic range are many and complex. Aerodynamic heating resulting from greater speeds continues to add a host of new problems and to complicate those of a long standing nature further. The need for increased research in the field of aircraft construction is evident.

The NACA, during the past year, has continued its efforts on the important problems associated with structural strength, efficiency, loading, flutter, fatigue, and materials under normal temperatures. It has also developed research tools and techniques for investigating aircraft under the elevated-temperature conditions encountered in high-speed flight. Further, it has succeeded in defining and exposing new thermal problems which future high-speed aircraft will encounter and has found solutions to certain of these problems.

Most of this research has been performed at the NACA laboratories with additional assistance provided by educational and other nonprofit institutions under contract to the NACA. A description of the Committee's recent unclassified research in the field of aircraft construction is given in the following pages and is divided into four sections: (1) Aircraft Structures; (2) Aircraft Loads; (3) Vibration and Flutter; and (4) Aircraft Structural Materials.

AIRCRAFT STRUCTURES

Static Properties

The use of integrally stiffened skins on aircraft is increasing because of the possibilities of saving weight and eliminating rivets and bolts. Compared with riveted-on stiffeners, integral stiffeners participate more fully with the skin in resisting external loads but, because of this action, may lead to an undesirable coupling of plate distortions for certain proportions and loading conditions. The nature of this problem is discussed in Technical Note 3646 where the modifications to the equations for stress distribution and deflection are made to account for the effects of coupling. Conditions under which the effects of coupling are significant are given in this paper.

Because engineering beam theory fails for deflection analysis of thin low-aspect-ratio wings, the development of efficient methods of analysis has become a problem. A matrix method based on energy principles for obtaining influence coefficients is presented in Technical Note 3640. The required matrices may be set directly from the data of the wing design. The necessary calculations have been arranged to take full advantage of automatic computing machines.

The thick-skin multiweb box beam is representative of wings of high-speed aircraft. Experimental data and strength analysis of this component are presented in Technical Note 3633. The combinations of design parameters which lead to minimum structural weight for various values of a loading index are given. The results are presented in such a manner that the lightest weight structure which satisfies wing-stiffness requirements can be found.

Classical theories of the structural strength and stability of plates assume that the plate deflections experienced are small in comparison with the plate thickness. In order to evaluate the inaccuracies resulting when this assumption is not fulfilled, Columbia University has developed a nonlinear plate theory of motion and solved the equations for certain dynamic cases. Underlying assumptions of various plate equations have also been studied. The results of this study are presented in Technical Note 3578.

Comparisons between the results of a theory for calculating stresses around cutouts in stiffened cylinders and the results of experiment are presented in Technical Note 3544. The data and the theory were previously published and coefficients for use with the theory have been calculated and published in Technical Note 3460. The theory takes into account the bending flexibility of the ring stiffeners. The comparisons show that good agreement is obtained if this factor is correctly accounted for.

New York University has conducted, under NACA sponsorship, a critical review of the literature published since 1940 on buckling and failure of plate elements. The results of this review, including a compilation of existing theories and experimental data, are presented in Technical Note 3781. A similar review has also been made at New York University of the existing literature on buckling of composite elements. The results of this review are presented in Technical Note 3782. During these reviews, general equations for the plastic buckling of cylinders were derived. These equations were then used to obtain solutions for the compressive and torsional buckling of long cylinders in the plastic region. These results, as well as comparisons between computed and test data, are presented in Technical Note 3726.

An analysis of the stresses in the plastic range around a circular hole in a plate was made both to explore means for solving stress problems in the plastic range and to obtain the solution of this basic problem. The results are presented in Technical Note 3542. Calculations were made for four different materials and the resulting stress-concentration factors are compared with those derived from a previously developed approximate formula.

Dynamic Properties

The major role that flutter plays in the design of high-performance aircraft requires that methods for computing accurate vibration modes and frequencies be obtained. In Technical Note 3636, the investigation of the usefulness of the substitute-stringer method for including the effects of shear lag in the calculation of the transverse modes and frequencies of box beams is continued. Box beams, the covers of which consist of normal-stress-carrying stringers on sheets carrying not only shear but also normal stress, are analyzed exactly. Frequencies of beams with various numbers of stringers, obtained by means of this exact analysis, serve to determine the possible accuracy of the frequencies obtained by the substitute-stringer approach. A combined experimental and theoretical investigation of the modes and frequencies of a large-scale built-up box beam is reported in Technical Note 3618. For bending vibrations, frequencies obtained from an analysis of a substitute-stringer structure which includes the influence of transverse shear deformation and shear lag were found to agree very well with those obtained experimentally. In the case of torsional vibrations, the frequencies obtained from either an elementary or a four-flange beam analysis which includes the effects of restraint of warping were found to be in satisfactory agreement with the experimental frequencies.

The vibration characteristics of hollow thin-walled rectangular beams have been investigated to obtain insight into the factors affecting the modes and frequencies of wings. The experimental results from this study are presented in Technical Note 3463 and indicate that the effect of shear deformation of the cross section on the torsional frequencies can be large. Further evaluation of this effect has been made and is presented in Technical Note 3464.

Thermal Properties

Rapid changes in temperature of the surface of an aircraft can induce thermal loads in the primary structure which may have serious aerodynamic and structural consequences. The nature of this problem was investigated by subjecting box beams which simulate high-speed-wing structure to a high-intensity heat source. These tests are reported in Technical Note 3474. It was found that the internal structure of the beams provided enough restraint against expansion of the heated skin surfaces to cause severe buckling of the skin. Buckling of the shear webs occurred during the cooling phase of the test when the temperature of the internal structure exceeded that of the skin. Measured strains were used to determine distortions and stresses which were found to agree with a thermal stress analysis of the test conditions.

One of the most important structural problems resulting from aerodynamic heating is the deterioration of material properties at elevated temperatures. This deterioration of material properties produces loss of strength and creep of structures and can lead to weight increases that adversely affect the performance of high-speed aircraft. A study has been made of the strength and creep behavior of aircraft structural elements at elevated temperatures to obtain methods for predicting structural behavior from material characteristics. One of these studies, reported in Technical Note 3552, was concerned with the elevated-temperature compressive strength and creep lifetime of simply supported plates. A similar study on the compressive strength and creep lifetime of skin-stringer panels is reported in Technical Note 3647. Both studies indicate that elevated-temperature strength of structural elements can be predicted from methods available for determining room-temperature strength provided that the appropriate stress-strain curve for elevated-temperature material is used. Previously reported studies of the elevated-temperature buckling strength of structural components have indicated similar results. The present studies also show that creep lifetime of structural elements may be determined from methods used to determine structural strength if the compressive creep properties of the material are substituted for the material stress-strain curve. The results make it possible to estimate the effect of creep on the weight of structures that are designed to operate at elevated temperatures.

The transient thermal stresses produced by aerodynamic heating of supersonic aircraft depend upon the temperature distribution within the structure, which, in turn, can be markedly influenced by the thermal conductivity of any joints present. In order to investigate the effects of joint conductivity on the thermal stresses in aerodynamically heated skin-stiffener combinations under various aerodynamic conditions, a theoretical study was made. In this study an aerodynamic heat-transfer parameter (called the Biot number), a joint-conductivity parameter, and geometrical proportions were varied. The results, presented in Technical Note 3699, indicate that increasing the joint conductivity beyond a certain value results in almost no change in the maximum skin or stiffener stresses; but, as the joint conductivity approaches zero, the maximum skin and stiffener stresses increase appreciably. Increasing the Biot number, an index of the rate of transfer of external heat to internal heat, can also cause a considerable increase in the maximum skin and stiffener stresses. However, when the Biot number is large (high rate of external heating), the value of the joint conductivity is relatively unimportant since the structure is heated so fast that there is no time for heat to be conducted into the interior of the structure; the joint conductivity thus affects the thermal stresses most significantly when the external heating rate is low. Changing the geometric characteristics produces results which are essentially independent of the joint conductivity and the Biot number.

In the design of aircraft structures, where aerodynamic heating is encountered, knowledge of the temperature distribution within the structure is of considerable importance. Because interior elements of the structure are heated by conduction through joints, the influence of various joint properties on thermal conductance has been investigated previously and reported by Syracuse University. Before extending this investigation, Syracuse University explored the influence of joint conductance on the transient temperature distribution in composite aircraft joints. Fabricated specimens representative of typical skin-stringer cross sections, as well as geometrically similar specimens without joints, were tested under aerodynamic heating conditions and the results from the two sets of joints were compared. The results, which are presented in Technical Note 3824, indicate that, in the practical case, joint conductance must be taken into account if temperature distributions throughout composite structures are to be predicted accurately.

Aircraft structures for high-speed flight must be designed so that excessive creep deformation and creep rupture does not occur during the design lifetime of the structure. An understanding of the creep behavior of structures is therefore necessary in order to eliminate such failures. A previously reported investigation by the National Bureau of Standards indicated that creep deformations within joints may be responsible for a considerable portion of the overall deformation of structures. However, no correlation was obtained between the creep of a riveted joint and the creep of its component materials. This study has now been extended and creep-test results of a number of additional joints are reported in Technical Note 3842. Methods are presented by which the time to rupture, the mode of rupture, and the deformation of structural joints in creep may be predicted. These methods are based upon the creep properties of the materials of the joint in tension, shear and bearing.

Aircraft structural elements subjected to long periods of heating and compressive loadings can buckle even though the applied load is less than the critical load of the element at the elevated temperature. This phenomenon is called creep buckling. Research equipment and techniques have been developed at the Polytechnic Institute of Brooklyn and are presented in Technical Note 3493. Additional creep-buckling tests of 2024T–4 aluminum alloy columns besides those published in this report have been conducted and the results correlated with theory.

The aircraft designer at the present time must deal with a multiplicity of materials and material properties which vary with temperature. It is essential, therefore, that speedy and accurate methods for predicting the influence of changes in material properties on structural strength be available. Such methods are given in Technical Note 3553 and Technical Note 3600 for various types of structural components which fail by compressive crippling. The methods utilize the concept of crippling-strength moduli which are readily calculated from the compressive properties of the material in the structure. Accuracy of the methods is illustrated with experimental data obtained in various materials and under different temperature conditions.

The transient temperature distributions produced by aerodynamic heating of thin solid wings induce thermal stresses that may effectively reduce the stiffness of the wing. This is a new problem that can be a significant factor in the aero-elastic behavior of aircraft structures. Such reductions in stiffness have been investigated experimentally by rapidly heating the edges of a cantilever plate. The midplane thermal stresses imposed by the nonuniform temperature distribution caused the plate to buckle torsionally, increased the deformations of the plate under a constant applied torque, and reduced the frequency of the first two natural modes of vibration. Small-deflection plate theory, employing energy methods, predicted the general effects of the thermal stresses but became inadequate when plate deflections were large. Additional studies have been initiated to investigate these effects.

AIRCRAFT LOADS

Basic Load Distribution

Extensive flight investigations have been made with the X–5 variable-wing-sweep research airplane at Mach numbers up to 1.0 to determine the effects on the wing and horizontal tail loads of varying the angle of wing sweep without modifying the other characteristics of the airplane. Up to a Mach number of 0.85, the balancing horizontal-tail loads measured in flight show a consistent variation as the wing sweep angle is increased from 20° to 59° with the greatest down tail load occurring at sweep angles of about 36°. The wing loads were found to have a nonlinear variation with airplane angle of attack and to reflect the changes that occurred in the wing characteristics. In another flight investigation, pressure measurements over the midspan station of the 8-percent-thick wing on the X–1 airplane in the transonic speed range showed a rearward movement of the chordwise load center with increasing Mach number with a particularly rapid and large movement in the Mach number range of 0.82 to 0.88. In the Mach number range 0.95 to 1.25 at high normal force coefficients, upper surface pressure distributions approached a rectangular slope.

In Technical Note 3476, spanwise lift distributions have been calculated for 61 swept wings with various aspect and taper ratios and a variety of angle-of-attack distributions including flap and aileron deflections. The information presented can be used both in the analysis of untwisted wings or wings with known twist distributions and in aeroelastic calculations involving initially unknown twist distributions. The information presented in Technical Note 3476 supplements similar information previously given in Technical Note 3014 for unswept wings so that the two papers cover all practical plan forms.

A method for computing the span loads and the resulting rolling moments for sideslipping wings of arbitrary plan form in incompressible flow is presented in Technical Note 3605. The basic method requires mechanical differentiation and integration to obtain the rolling moment for a wing of arbitrary plan form in sideslip when the span load at zero sideslip is known The mechanical differentiation and integration can be avoided, however, by use of a step-load method which is also derived. A comparison of the calculated span loads and rolling-moment parameters with available experimental data shows good agreement.

The development of new-type control devices requires that structural design data be provided. The effects, therefore, on the chordwise pressures and section forces and moment coefficients near midspan of deflecting various plain spoilers and a flap-type control with and without an attached tab on a swept wing have been investigated at Mach numbers from 0.60 to 0.93.

In order to design aircraft one must have a knowledge of body effects on the wing spanwise load distribution at all speeds. Although methods exist for predicting such body effects on sweptback wings at low speeds, practically no direct experimental verifications have been available. In a recent investigation, detailed wing pressure-distribution data that permit the desired comparison were obtained. The data, reported in Technical Note 3730, indicated that, although previous methods did not satisfactorily predict body effects on the unflapped uncambered wing, a swept-wing method employing 19 spanwise lifting elements and control points gave good agreement except when the wing had deflected trailing-edge flaps or was cambered and twisted.

Normal-force and normal-pressure distributions for an ogive-cylinder body of revolution of fineness ratio 10 are reported in Technical Note 3716 for a free-stream Mach number of 1.98 and an angle-of-attack range from 0° to 20°. Comparisons of experimental and theoretical normal-force and normal-pressure distributions indicate that available theoretical methods can be used to predict experimental results with good accuracy for angles of attack to about only 5°. At greater angles of attack, the normal-force distributions differ significantly from those calculated in accordance with theories which include methods of estimating the effects of viscosity on the forces and moments for inclined bodies. Analysis of the data shows that these differences are, in general, attributable to inadequate estimates of the magnitude and distribution of the cross forces resulting from flow separation. A correlation curve for the longitudinal distribution of the cross-flow drag coefficient for laminar boundary-layer flow has been developed and is based upon the assumption that the distribution depends primarily upon the body shape. It is believed that use of this curve for the viscous cross-force contribution in conjunction with first-order linear theory for the potential cross force provides a satisfactory method for estimating normal-force and pitching-moment characteristics for similarly shaped bodies with laminar-boundary-layer flow.

In Technical Note 3479, horizontal-tail loads measured in gradual and abrupt longitudinal maneuvers on two configurations of a four-engine jet bomber are presented. The least-squares procedures were used to determine aerodynamic loads from strain-gage measurements of structural loads. The results are analyzed to determine the flight values of the aerodynamic coefficients which are important in calculations of horizontal-tail loads for comparison with wind-tunnel results. The effects of fuselage flexibility on the loads are determined and some calculations of critical horizontal-tail loads beyond the range of the tests are compared with the design loads.

Some indication of the importance of the directional-stability characteristics of present-day high-speed airplanes with increasing angle of attack and Mach number has become apparent from recent wind-tunnel tests. An analysis of wind-tunnel data has shown that the vorticity shed from the nose of the fuselage and directed by the wing to strategic locations in the vicinity of the vertical tail markedly affects the load on the vertical tail in sideslip at high angles of attack and supersonic Mach numbers. For such conditions, the directional stability of the airplane may become negative.

Gust Loads

The collection of data with NACA VG and VGH recorders to determine the magnitude and frequency of occurrence of the gusts and gust loads and the operating air speeds and altitudes of commercial transport airplanes has been continued. The VGH data covering about 3,000 hours of operation from two types of four-engine transport airplanes currently in use on transcontinental and eastern United States routes are presented in Technical Notes 3475 and 3483. The analysis of these data indicates that the more severe gust loads occurred for operations over the eastern portion of the United States, a result attributable to the higher operating speeds in rough air for these operations. A related study of approximately 70,000 hours of VG data from six different operations of twin-engine transport airplanes over the past eight or nine years, presented in Technical Note 3621, indicates that the loads and gusts were comparable with those experienced in previous operations of the same type of airplane.

The information available on the spectrum of atmospheric turbulence is briefly reviewed in Technical Note

3540 and a method is presented for converting available gust statistics normally given in terms of counts of gust peaks into a form appropriate for use in spectral calculations. The fundamental quantity for this purpose appears to be the probability distribution of the root-mean-square gust velocity. Estimates of the variation of this distribution with altitude and weather condition are also derived from available gust statistics. A critical problem in connection with the design and operation of missiles and airplanes capable of high-speed vertical flight arises from the loads and motions experienced when intense layers of wind shear are encountered. As a consequence, data on the magnitude and frequency of occurrence of the shear layers at different altitudes and seasons were determined from U. S. Weather Bureau rawinsonde data and are reported in Technical Note 3732. These data indicate that maximum shear intensities of about 120 feet per second per 1,000 feet occur at altitudes of about 50,000 feet during the spring and winter seasons but occur in relatively thin layers having thicknesses not greater than about 3,000 feet.

A method for obtaining a power spectrum of vertical gust velocity over a wide range of wave length has been devised and test results are published in Technical Note 3702. A spectrum of vertical gust velocity was measured at low altitude in clear-air turbulence having a root-mean-square intensity of 5 feet per second for wave lengths from 10 feet to 60,000 feet. At the higher frequencies (short wave lengths), the power spectral density varied at a rate which was approximately predicted by theory. The spectrum which was obtained tended to flatten out for the longest test wave lengths. The break frequency which provides an indication of the scale of the turbulence occurred at a wave length of approximately 6,000 feet.

Calculated unsteady-lift functions and spanwise lift distributions for delta, rectangular, and elliptical wings undergoing a sudden change in sinking speed are presented in Technical Note 3639. These data indicate that the normalized unsteady-lift functions are substantially independent of the plan form for elliptical, rectangular, or moderately tapered wings, but for delta wings the increase of lift toward the steady-state value is much more rapid. The results in this report corroborate the results of other investigations which show that the rate of growth of lift tends to increase with a decrease in aspect ratio and that spanwise distributions of the indicial lift seem to be independent of time for rectangular and elliptical wings. In Technical Note 3748, reciprocal relations for unsteady flow are used to calculate total-lift responses of wings to sinusoidal gusts and to sinusoidal vertical oscillations. A variety of plan forms are considered for incompressible, subsonic compressible, sonic, and supersonic flow. A theory is presented in Technical Note 3805 for calculating the variation with frequency of the lateral-force and yawing-moment coefficients due to sinusoidal side gusts passing over the profile of a simple fuselage combined with a vertical fin. Since slender-body theory is used, the results are applicable to both subsonic and supersonic airspeeds, provided the local flow angles between the profile and the airstream are small.

An investigation to determine the gust-alleviation capabilities of fixed spoilers and deflectors on a transport-airplane model incorporating a straight wing is reported in Technical Note 3705. The results indicate about equal effectiveness (from 20 to 40 percent) of spoilers or deflectors in reducing normal accelerations in rough air through reductions in lift-curve slope. Both devices were also equally effective in decreasing the airspeed through increased drag. In Technical Note 3746, the wing and horizontal-tail loads and spar strains measured on a twin-engine light transport airplane, modified by a gust-alleviating device for passenger comfort, were presented. The results presented are an initial analysis of samples of measurements obtained in clear-air turbulence with the alleviation system both off and on. Although the alleviation system was not optimum, the root-mean-square normal acceleration at the airplane center of gravity was reduced by 43 percent and the wing bending strains were reduced, but wing-shear strains and horizontal-tail shear and bending strains were increased.

Landing Loads

In Technical Note 3541, a method is presented for statistically deriving contact vertical velocities of airplanes from measurements of maximum incremental center-of-gravity acceleration at contact. Probability curves of derived velocities for a test airplane when compared with curves of measured velocities show a difference of less than 0.2 foot per second throughout the velocity range covered in the investigation. A statistical comparison of the landing-impact velocities of the first and second wheel to touch ground from about 350 transport landings is reported in Technical Note 3610. The comparison indicates that the mean vertical velocity at the instant of contact was about the same for either wheel but that the probability of a high value of vertical velocity was somewhat greater for the second wheel to touch than for the first. The effect of the rolling velocity of the airplanes at the instant of initial contact was to increase the vertical velocity of impact of the wheel toward which the airplane was rolling regardless of whether it was the first or second wheel to touch. There appeared to be no definite influence of the ratio of landing-gear tread to radius of gyration of the airplanes on the relative vertical velocities of the first and second wheels to touch, as would be expected from theoretical considerations.

Technical Note 3604 reports results of tests made to determine the lateral or cornering force, drag force, torsional moment or self-alining torque, pneumatic caster, vertical tire deflection, lateral tire deflection, wheel torsion or yaw angle, rolling radius, relaxation length, tire footprint area, and variation of unloaded tire radius with inflation pressure for two 26- by 6.6-inch, type VII, 12-ply-rating tires. Data were recorded for conditions of rectilinear-yawed rolling over a range of inflation pressures and yaw angles at the rated vertical load and at twice the rated vertical load. Vibration tests were made to determine the dynamic lateral elastic characteristics of the tires. During rectilinear-yawed rolling, the normal force generally increased with increasing yaw angle within the test range, the variation of normal force with yaw angle differed for the two vertical loads tested, the pneumatic caster was a maximum at small yaw angles and tended to decrease with increasing yaw angle, and the sliding drag coefficient of friction tended to decrease with increasing bearing pressure.

A comprehensive correlation, evaluation, and extension of linearized theories for tire motion and wheel shimmy has been made and is reported in Technical Note 3632. It is demonstrated that most of the previously published theories represent varying degrees of approximation to a summary theory developed therein which is a minor modification of the basic theory of Von Schlippe and Dietrich. In most cases where strong differences exist between the previously published theories and the summary theory, the previously published theories are shown to possess certain deficiencies. Comparison of the existing experimental data with the predictions of the summary theory provides a fair substantiation. Some discrepancies exist however, which may be due to tire hysteresis effects or other unknown influences.

Theory indicates a sharp increase in the hydrodynamic load as the dead-rise angle approaches zero. There have been, however, few experimental data available for verifying the loads predicted by theory for angles of dead rise below 20°. Results of a brief investigation of the loads in smooth water for 10° angle of dead rise are reported in Technical Note 3608 and are compared with theory for immersed hydrodynamic impact of nonchine bodies. The trend of the experimental variation of load-factor coefficient, draft coefficient, time coefficient and velocity ratio is in good agreement with the theoretical variation.

Technical Note 3619 presents data showing the effect of horizontal restraint of carriage mass in experimental testing facilities upon the general theoretical equations of motion for the prismatic body during a hydrodynamic impact. The data indicate that the carriage mass has little effect for the low trims, since at this condition the resisting water force has only a small component in the horizontal direction, but for the higher trims the effect is appreciable. For the more usual seaplane-design conditions, that is, approach parameters larger than 1.0 and trims up to 15°, the maximum correction for any of the coefficients is 10 percent or less.

Research Techniques

It is frequently desirable to predict the loads that would be experienced with more hazardous control motions or flight conditions than those for which test data exist. Accordingly, considerable effort has been expended in developing and comparing various methods by which such predictions can be made. Fourier and Laplace transforms and the type of analyses used in studies of servomechanisms have been used extensively in this development. It appears from the work accomplished that the concept and use of a unit impulse as a research technique has considerable merit. Simple and rapid methods for determining the time response to a unit impulse from frequency-response data and for evaluating the Fourier transform as a function of time have been derived and are presented in Technical Note 3598. These methods are applicable to linear functions for which Fourier transforms exist, which is usually the case in the treatment of airplane maneuvers. In Technical Note 3701, the method developed in Technical Note 3598 is compared with several other methods of obtaining the time response of linear systems to either a unit impulse or to an arbitrary input from frequency-response data. The comparisons indicate that most methods gave good accuracy when applied to a second-order system; the main difference is in the computing time. In general, the method of Technical Note 3598 was advantageous in all respects, since it was more accurate and required less time.

VIBRATION AND FLUTTER

Flutter

The sonic and supersonic speeds of modern aircraft plus their use of relatively flexible thin wings and stabilizers have caused flutter to assume a more important role in aircraft design. In addition to research on the flutter characteristics of typical aircraft configurations, research is also being carried out to understand better the aerodynamic, structural, and inertial considerations inherent in flutter.

On the basis of an analysis of a large quantity of flutter data taken from subsonic, transonic, and supersonic wind tunnels and from rocket- and bomb-drop tests for a wide variety of wing plan forms, a criterion was derived which permits a rapid estimate of the probability of flutter for lifting surfaces. This criterion groups the significant parameters into simple geometric dimensions and structural properties. Another simple criterion was developed for stall flutter.

A number of swept wings having systematic variations in plan form and structural characteristics have been flutter tested in transonic and supersonic wind tunnels to establish the effect of various parameters on flutter and to serve as a basis for evaluating analytical procedures. Because of the large number of parameters involved, this is a large test program and is still underway.

An alternative to the testing of a systematic series of wind-tunnel models in order to establish the influence on flutter of elastic and inertial structural characteristics is to employ an analog computer whose electrical elements and behavior approximate the elements and dynamic behavior of the structure. Such an analog study has been carried out at the California Institute of Technology and is discussed in Technical Note 3780. Four wings representative of those of current aircraft were considered and the effects of changes in bending and torsional stiffnesses, mass distribution and angle of sweep on the flutter characteristics were determined. A sufficient number of cases were treated to establish the trend over a sizeable range for each parameter.

As reported in the Forty-First Annual Report, 1955, a theoretical study of the flutter of two-dimensional panels was reported in Technical Note 3465. More recently, flutter of panels mounted on the wall of a supersonic wind tunnel was obtained at a Mach number of 1.3. It was found that, at the flow conditions of these tests, increasing the tensile forces in the panel was effective in eliminating flutter, as was shortening the panels or increasing their bending stiffness. No apparent systematic trends in the flutter modes or frequencies could be observed, and it is significant that the panel flutter sometimes involved higher modes and frequencies. The presence of a pressure differential between the two surfaces of a panel was observed to have a stabilizing effect. Initially buckled panels were more susceptible to flutter than panels without buckling. Buckled panels with all four edges clamped were less liable to flutter than buckled panels clamped only on the front and rear edges.

In Technical Note 3638, a preliminary theoretical investigation of the panel flutter and divergence of infinitely long, unstiffened and ring-stiffened, thin-walled, circular cylinders is described. Linearized unsteady potential-flow theory was utilized in conjunction with Donnell's cylinder theory to obtain equilibrium equations for panel flutter. Where necessary, a simplified version of Flügge's cylinder theory was used to obtain greater accuracy. By applying Nyquist diagram techniques, analytical criteria for the location of stability boundaries were derived. This report also includes a limited number of computed results.

One of the most troublesome types of flutter is that involving oscillations of a control surface at transonic speeds, commonly referred to as buzz. In Technical Note 3687, results of wind-tunnel tests of three wing models are presented and it is shown that a large range of change in density of the test medium had little effect on the initial magnitude and initial Mach number of buzz. The buzz frequency decreased somewhat with decrease in density. The Mach number corresponding to the onset of buzz decreased as the wing angle of attack was increased. Mass balance and changes in spring stiffness changed only the oscillation frequency. The test results indicated that placing the aileron at the wing tip delayed the onset of buzz to higher Mach numbers. A comparison of the experimental results with two published empirical analyses showed only qualitative agreement.

Designers of thin aircraft wings must consider the possibility of wing torsion flutter at high angles of attack, which is referred to as stall flutter. The results of an exploratory, analytical, and experimental study of some of the factors which might be of importance in the stall-flutter characteristics of thin wings are prepresented in Technical Note 3622. The factors considered were Mach number, Reynolds number, density, aspect ratio, sweepback, structural damping, location of the torsion node line, and presence of concentrated tip weights. The importance of aerodynamic torsional damping on the stall flutter of thin wings was demonstrated by comparison of the regions of negative torsional damping measured on a spring-mounted model with the regions of flutter. The results of a series of experiments on a thin wing tested at various spans indicated that compressibility alters the stall-flutter characteristics and that these effects depend upon aspect ratio. A brief study of the inertia effects of concentrated weights at the tip indicated that such effects can be important. An approximate analysis is presented for such configurations.

Aerodynamics of Flutter

It has been demonstrated that generalized forces for a harmonically oscillating wing in pure supersonic flow may be expressed in terms of certain integrals commonly referred to as f_λ functions. These functions have been tabulated on a large computer for a wide range of parameters important to flutter and the tabulated results are presented in Technical Note 3606.

A fundamental study of the aerodynamic forces on an oscillating wing is presented in Technical Note 3643. This report presents the magnitude and phase angle of the components of normal force and pitching moment acting on an airfoil oscillating in pitch about the midchord at both high and low mean angles of attack and for Mach numbers of 0.35 and 0.70. The magnitudes of normal-force and pitching-moment coefficients were much higher at high mean angles of attack than at low angles of attack for some conditions. Large regions of angle of attack and reduced frequency were found wherein one-degree-of-freedom torsion flutter is possible. It was shown that the effect of increasing the Mach number from 0.35 to 0.70 was to decrease the initial angle of attack at which unstable damping occurred. In addition, the aerodynamic damping in essentially the first bending mode was measured for two finite-span, 3- and 10-percent-thick wings for a range of mean angles of attack and reduced frequencies. No regions of negative damping were found for this motion, and it was found that the damping measured at high angles of attack was generally larger than that at low angles of attack.

An experimental study of the lift and moment about the quarter chord of an oscillating wing at high subsonic Mach numbers is presented in Technical Note 3686. A comparison of the experimental magnitude of the lift vector with the theory as given by Dietze showed good agreement. Comparisons with theory of the moments and the out-of-phase component of lift indicated that some refinements in the testing technique are necessary for the experimental determination of these quantities in the transonic speed range.

An experimental wind-tunnel investigation was carried out of the forces, moments, and phase angles on a two-dimensional wing equipped with an oscillating circular-arc spoiler. Schlieren photographs were obtained which showed the flow over and behind the spoiler while it was oscillating. The forces and moments on the wing were obtained from instantaneous pressure-distribution measurements. The results indicated that the effects of Reynolds number on the normal-force and moment coefficients and their phase angles were very small and somewhat erratic. An increase in Mach number increased the normal-force coefficient and had no consistent effect on the moment coefficient, while the phase lag of both the normal force and moment decreased. There was little effect of reduced frequency on the normal-force coefficient; however, increasing the reduced frequency produced an essentially linear increase in the phase lag of the normal force.

Buffeting

Several studies have been made of the available transonic Mach number data on wing dropping, low-lift buffeting, buffet boundaries, and changes in the angle of zero lift for symmetrical airfoils and various airplane configurations. These phenomena are indicated to be allied and are probably the result of shock-induced separated flow. It was found that unswept wings which have airfoil sections 9 percent thick or thicker are susceptible to wing dropping at transonic speeds. Wing dropping may occur even for thin wings, however, if the airfoil contour is not fair. Sweepback only partially relieves the wing dropping and buffeting problem for thick wings. The studies have also indicated that there are combinations of airfoil-thickness ratio, aspect ratio, and sweep angle which may allow flight through the transonic speed range without either wing dropping or buffeting at low lift. Decreases in aspect ratio and thickness ratio and increases in sweepback all tend to alleviate high-speed buffeting. Low-lift buffeting, however, may be induced by the interference effects of thin intersecting surfaces such as a tail arrangement in which the horizontal tail is mounted above the fuselage on the vertical tail. Such a tail arrangement may also be partially responsible for large transonic trim changes and may exhibit an increase in drag over that for a comparable tail arrangement where the horizontal tail is mounted on the fuselage.

An analysis of some statistical properties of the buffet loads measured on the unswept wing and tail of a fighter airplane has indicated that buffeting can be considered as a random process. Buffet loads measured on the wing and tail in both the stall and shock regimes indicated that the wing loads in buffeting can be treated as the response of a simple elastic system to a random input. The wing buffet loads were normally distributed and the probability that a peak load would exceed a given level was in agreement with theoretical results. There was evidence that the tail buffet loads were not normally distributed as the wing loads but appeared to represent a more complicated process. The spectrum of the wing-root shear indicated that the buffet loads were primarily associated with response in first symmetrical bending. The spectrum for the tail-root shear indicated that the tail buffet loads were associated with the fuselage-torsion or tail-rocking mode. This study was reported in Technical Note 3733.

AIRCRAFT STRUCTURAL MATERIALS

Structural Materials at High Temperatures

Aerodynamic heating continues to be the source of the most perplexing and urgent problems in the field of aircraft structural materials. This is true in extreme cases such as long-range ballistic missiles where the severity of the requirements will clearly necessitate the development of new kinds of structural materials and new kinds of test facilities. In addition, it is true for less severe applications such as manned airplanes, where the effects of high temperatures on the common engineering properties of existing materials are so inadequately known that the designer lacks the handbook data he needs to arrive at an efficient, yet safe, design. In effect, heat has introduced a new dimension in all material problems; strain-rate effects, changes in modulus, creep, stress rupture, thermal stress, thermal conductivity, and many other temperature-linked characteristics, which heretofore could be ignored, will have to be taken into consideration in the future. Some of these problems are under attack on several fronts.

The tensile properties of a number of structural materials under rapid-heating conditions were determined by means of a new type of test (a so-called rapid-heating test) in which the material is first loaded and then heated at various heating rates until yield and failure occur. Sheet materials used in this investigation included 7075–T6 and 2024–T3 aluminum alloys (Technical Note 3462, reported in the Forty-first, 1955, Annual Report), Inconel and RS–120 titanium alloy (Technical Note 3731), HK31XA–H24 magnesium alloy (Technical Note 3742), and AZ31A–O magnesium alloy (Technical Note 3752). In these tests, heating rates have been varied from 0.2° to 100°F per second. At the higher heating rates, the materials were found to be stronger, in general, than under constant-temperature conditions when loaded at a strain rate of 0.002 per minute. In most cases, yield stress, rupture stress, and temperature have been found to be correlated by means of a temperature-rate parameter. Some of the materials, such as the new high-temperature magnesium alloy HK31XA–H24, exhibited a marked increase in strength at high heating rates in the high-temperature region. Other materials, such as 2024–T3 aluminum alloy and RS–120 titanium alloy, behaved in a very complicated manner under rapid heating.

In an investigation, conducted at the University of Alabama under NACA sponsorship, the fatigue strengths at 10 million cycles of two of the more promising titanium alloys, 3Mn Complex and 3Al–5Cr, were determined at 200°, 400°, 600°, 800°, and 1,000°F. Data of this sort are needed for the evaluation of these new alloys of titanium before the role they can play in the solution of some phases of the high-temperature problem can be predicted.

The use of thermal insulation on the surface of structural materials is one of several possible methods of defeating the adverse effects of aerodynamically generated heat. However, there are many fundamental and technological difficulties, such as the realization of adequate strength of the coating-to-metal bond, which stand in the way of achieving practicable coatings. Results of an investigation conducted at the National Bureau of Standards and reported in Technical Note 3679 show that copper ions in the coating have the effect of producing a significant increase in the adhesion between the coating and the surface of stainless steel.

Laminates of nonmetallic materials possess characteristics which uniquely suit them for use in certain components of aircraft. The rate of deterioration of their mechanical properties with temperature, however, is a deterrent to their use in very fast aircraft. Future progress demands that improved materials be developed and further test data be obtained to enable the designer to gage the range of applicability of existing laminates. In an investigation conducted at the University of Illinois and reported in Technical Note 3414 the static-tension, static-compression, tension-creep, and time-to-fracture characteristics of melamine-resin glass-fabric laminates and silicone-resin glass-fabric laminates at temperatures up to 600°F were determined. In the analysis of the creep data an equation based on the activation-energy theory, which describes the effects of stress, time, and temperature is reported.

Fatigue

Failure by fatigue has always been and still is a potential hazard in aircraft structures and is therefore an important subject for research. Although steady and significant progress has been made in understanding the phenomena of fatigue and in designing structures that will incorporate characteristics that both lessen the likelihood of fatigue cracks and preserve the integrity of the structure when a crack does develop, there are still many aspects of the fatigue problem that require solution. Among these is the stress-concentration effect of geometrical discontinuities on fatigue properties of aircraft structural materials. Technical Note 3631 presents the results of axial-load fatigue tests on 2024–T3 and 7075–T6 aluminum-alloy sheet specimens with central holes. Specimens with various combinations of hole diameters and widths were tested to provide data suitable for study of the geometrical size effect.

In Technical Note 3293, which reports an investigation conducted at the National Bureau of Standards, the results of cumulative-damage tests of 7075S–T6 and 2024S–T3 aluminum-alloy sheets under various loading conditions are given. The cumulative damage ratio, which should be unity if the theory were absolutely correct, was found to vary from 0.568 to 1.440; however, 40 percent of the cumulative damage ratios were within 10 percent of unity.

At the University of California, a study (Technical Note 3495) was made of fatigue under combined repeated stresses with superimposed static stress. A comprehensive critical review of the literature where such tests were reported was made. In addition, tests were performed to determine the effects of static compression on alternating torsion, which was the only combination that had received no previous attention. The results were compared with the predictions of theory. It was shown that the Orowan theory of the effects of combined stress and cyclic stress on fatigue can be modified to predict the observed test results.

In an investigation conducted at the Battelle Memorial Institute, effects of notch severity on the initiation and propagation of fatigue cracks in ¼- and 2-inch-diameter notched bars were determined in rotating bending fatigue tests. These bars consisted of 2024S–T4 aluminum alloy with stress concentration factors of 5.2 and 13.9. The results reported in Technical Note 3685 indicate that cracks initiate in severely notched bars earlier than in unnotched or mildly notched bars. Discernible cracks occurred at 1,000 cycles at stress levels that would result in failure at 200,000 to 1,700,000 cycles. Differences in results of the tests of the ¼- and 2-inch-diameter bars indicated a size effect, which was attributed to residual stresses in the larger bars.

Fatigue stressing and the accumulation of damage have effects on the internal friction of metals and alloys.

Internal friction measurements are therefore useful in the study of the fundamentals of fatigue. In an investigation conducted at the California Institute of Technology and reported in Technical Note 3755, a correlation of internal friction and torsional fatigue was made at various temperatures. The results indicated the existence of a critical temperature at which fatigue life reached a minimum, and the effect of this temperature on internal friction was found to be substantial. In addition, the recovery of internal friction during periods of rest after fatigue stressing was observed. This recovery was found to be dependent on both stress level and temperature. Attempts were made to rationalize the relationship between the changes in the characteristics of internal friction and the inadequately understood phenomena of damage, recovery, and coaxing in fatigue.

Plastic Behavior of Metals

An understanding of the plasticity of metals is essential to the understanding of strength, ductility, resistance to brittle fracture, workability, and other properties of metals which account for their usefulness as aircraft structural materials. The NACA is conducting research in various areas of this field.

Technical Note 3681 reports results of an investigation conducted at the Battelle Memorial Institute on the plastic behavior of binary aluminum alloys by internal-friction methods. Effects of strain rate, amount of strain, heat treatment, temperature, and cyclic frequency on internal friction were determined, and the results were analyzed and rationalized in the framework of the dislocation theory of plasticity.

During plastic deformation of materials, distortions occur which are not predictable by the usual assumption of isotropy. Errors in strain of 50 percent and more resulting from anisotropy in the plastic range would not be uncommon for some of the materials used in aircraft construction. A series of tests, described in Technical Note 3736, are utilized to establish semi-empirical relationships between Poisson's ratio and the properties of the materials as shown by their stress-strain curves. The tests also show that there is no permanent change in volume of the metals tested after stressing into the plastic range.

In an investigation conducted at Battelle Memorial Institute and reported in Technical Note 3728, the structure of slip lines developed in single crystals of aluminum at various stages during tensile deformation were examined in an electron microscope. On the basis of experimental results from this work and others from the literature, a mechanism for slip-band formation based on dislocation theory was formulated. The possible effects of short-range ordering on deformation modes are discussed.

Non-Metallic Materials

Cotton fabric-phenolic laminates are useful structural materials for aircraft, but the knowledge of the effects of processing and manufacturing variables on their properties is inadequate. In an investigation conducted at the National Bureau of Standards and reported in Technical Note 2825, tests were conducted to determine strength properties of (1) several untreated commercial cotton fabric-phenolic sheet laminates, (2) the same materials after exposure to typical postforming heating cycles, (3) industrially postformed shapes made from one of these materials, (4) industrially-made and laboratory-molded shapes, and (5) flat panels postformed from the laboratory-molded shapes. It was shown that molding of phenolic laminates may or may not affect the strength, depending on the fabrication techniques used.

OPERATING PROBLEMS

During the past year, the NACA has continued to conduct research on various problems that are associated with the operation of today's modern high-speed aircraft. It is recognized that as the performance of the nation's aircraft is increased new problems are encountered and some old problems become more important in the day-to-day operation of these aircraft. Some of the most important current problems include the effect of aircraft noise on aircraft and people; atmospheric effects such as icing, turbulence, lightning, temperature, and density; and flight safety, which includes crash fire, survival, aircraft braking, visibility, engine reliability, foreign-object damage, and other problems. Working with the NACA Committee on Operating Problems are the Subcommittee on Aircraft Noise, the Subcommittee on Meteorology Problems, the Subcommittee on Icing Problems, and the Subcommittee on Flight Safety.

The effects of the intense noise from modern aircraft and missiles present one of the most serious problems which faces the civil and military aircraft operators today. This problem offers a great challenge to our technical ability to find a satisfactory solution. The NACA with the advice and active help of the Subcommittee on Aircraft Noise has expanded its research on noise with particular emphasis on understanding the mechanisms of noise production and noise suppression. The scope of the last several meetings of this Subcommittee has been expanded to be, in effect, limited conferences on aircraft noise and have had international participation. These meetings were arranged to provide a free and comprehensive discussion of the noise problems and research efforts of various groups active in this field. The results of such meetings have been profitable: A cooperative effort to utilize the intellectual resources and research facilities of all concerned with aircraft noise is essential to the solution of the noise problem.

As part of the constant NACA effort to summarize, discuss, and present recent NACA research results so that industry can best use these results for practical aircraft applications, technical conferences are held when appropriate for representatives of pertinent segments of the aircraft industry. On April 17, 1956, the NACA Conference on Airplane Crash-Impact Loads, Crash Injuries, and Principles of Seat Design for Crash Worthiness was held at the Lewis Flight Propulsion Laboratory. The NACA summarized the results of several years' work in the field of crash survival and proposed design criteria which, if applied, could improve aircraft seat design for crash survival. Civil and Military aircraft operators, manufacturers, and seat manufacturers now have the material presented at this conference as a guide for seat design. This material for the first time includes experimental data from actual dynamic-crash-load studies utilizing actual full-scale cargo, transport, and fighter-type aircraft.

A Summary of results of most of the recent unclassified investigations on operating problems is presented in the following paragraphs.

AIRCRAFT NOISE

The noises produced by current aircraft and missile power plants have increased to such intense levels that they affect the integrity of aircraft and missile structures, equipment, and control systems as well as present serious bioacoustic, efficiency, and annoyance problems for persons exposed to the noises. The effect of noises and related vibrations must now be considered as one of the principal elements of aircraft or missile design, and a specific NACA research program is directed toward obtaining full understanding and control of the production and effects of high-intensity noises.

While the noises produced by jet exhausts remain of primary concern to the NACA, research is also being conducted on boundary-layer and propeller noise, on the effects of noise on structures, and on propagation of noise through the atmosphere.

Jet Noise

NACA research has established that jet noise is produced by the turbulent mixing of the jet exhaust with the surrounding air; consequently, detailed investigations of the mixing phenomena are under way. One phase of this study consisted of measuring the turbulence in a subsonic jet by use of a hot-wire anemometer. The results are described in Technical Note 3561.

The studies begun last year of devices for altering the jet exhaust flow and thereby reducing jet noise are continuing. Tests on toothed and ejector nozzles are described in Technical Notes 3516 and 3573. Tests of the use of square, rectangular, and elliptical nozzles for subsonic jets, as reported in Technical Note 3590, showed that simple changes in jet-nozzle shape had very little effect on noise generation. It is shown in that report, however, that if the exiting flow is supersonic, a convergent-divergent nozzle operating near its design point will produce less noise than an ordinary convergent nozzle.

With the data presented in Technical Note 3591 for an investigation of the scaling parameters between various jet engines and model jets, it is possible to estimate the far noise field of a jet engine from its flow characteristics. Detailed data are also presented in that report for the noise field around a modern jet engine operating under static conditions with and without afterburner.

A method for limiting the noise received on the ground during takeoff of a jet aircraft is to control the operational techniques. A study of the effects of various climbing procedures, reported in Technical Note 3582, showed that lowest effective noise levels over the largest ground area will be obtained when the aircraft is climbing on the steepest flight path consistent with minimum safe airspeed.

Boundary Layer Noise

In addition to the noise problems caused by a jet exhaust, serious problems result from the noise produced by the boundary layer flow over the surface of the fuselage and wings. Preliminary flight tests have been made to determine the surface pressure fluctuations caused by a turbulent boundary layer. The relation of boundary-layer noise to Reynolds number, velocity, and altitude has been studied and further work is being done on flight at high subsonic velocities.

A study by the California Institute of Technology of subsonic and supersonic flow of air past rectangular cavities cut into a flat surface indicated that the cavities would emit a strong acoustic radiation. From that work as reported in Technical Note 3487 and the above NACA flight tests it appears that noise considerations may be a primary factor in establishing the limits for such items as surface finish and size and shape of surface cutouts or protuberances for high-speed aircraft.

As a part of its work on aerodynamic noises under NACA sponsorship, the California Institute of Technology developed an instrument of fairly simple design for measuring time correlation functions of two stationary random-input signals. The device and its use in determining auto-correlation functions are discussed in Technical Note 3682.

Propeller Noise

Instrumentation suitable for making flight measurements of the free-space sound pressures in the immediate vicinity of a propeller in forward flight has been developed and successfully used on a fighter airplane up to a Mach number of 0.72. The sensing element is a capacity microphone housed in a streamlined probe and used in conjunction with an oscillator to convert the pressure pulses into a frequency-modulated signal which is telemetered to the ground. At the ground receiving station, the telemetered signal is detected and recorded on magnetic tape. Subsequently, the recorded signal is converted to a varying voltage which is fed into a heterodyne frequency analyzer. This instrumentation is reported in Technical Note 3534.

Effects of Noise on Structures

NACA research is continuing on the problems of designing and constructing structures that will be suitable for use in the intense-noise-pressure fields near propellers and jets. The response of various structural systems to acoustic inputs, the stresses encountered in the systems, the fatigue characteristics of the systems, and the effects of insulation and damping on the structure are all under study. Stress data has been obtained for panels exposed to discrete and random noise levels of over 160 decibels. The fatigue life of panels was noted to decrease markedly for further nominal increases in the noise intensity level.[26] [27]

Attenuation of Noise

The NACA has continued its sponsorship of research at the Massachusetts Institute of Technology to determine the effects of terrain and atmospheric conditions on the attenuation of noise. A theoretical and experimental investigation of the sound field about a point source over a plane boundary in the presence of a vertical temperature gradient is reported in Technical Note 3494. Methods are presented for analyzing the effects of temperature gradients on the attenuation of sound in the shadow zone of a sound field. A further study has produced a semiempirical method for the calculation of a sound field about a source over ground. This study considered the effects of vertical temperature as well as wind gradients and the scattering of sound by turbulence into the shadow zone (Technical Note 3779).

A theoretical study of the sound field from a random noise source above ground as measured by a receiver with finite band width is presented in Technical Note 3557. It is shown that the far sound field still contains two major regions so far as attenuation over ground is concerned. In the first region, the sound pressure level decreases approximately 6 decibels per doubling of distance. In the second region, however, beyond a certain distance from the source, the level decreases monotonically 12 decibels per doubling of distance.

[26] See Regier paper listed on p. 78.
[27] See Lassiter, Hess, and Hubbard paper listed on p. 78.

FLIGHT SAFETY

During this second year of its existence the NACA Subcommittee on Flight Safety has not only monitored research into problems directly related to safety, such as fire, ditching, engine reliability, crash loads, and crash survival, but also studied results of research in other specialized fields so that they could be channeled directly to aircraft designers and operators through their safety organizations for immediate consideration. The following information shows the results of varied research projects that are significant for particular phases of aircraft operation which are considered to be most important from a safety standpoint.

Landing Problems

Operating statistics indicate that the landing phase of flight is most important from the standpoint of safety. There is much to be learned about the many facets of the landing problem. The NACA is actively studying many parts of the problem and has during the past year reported the results obtained with respect to landing loads, landing statistics, runway roughness and aircraft braking. In addition, the NACA has given wide distribution to the results obtained by other organizations investigating specific phases of the problem, namely nose-wheel shimmy (Technical Memorandum 1391) and friction of aircraft tires (Technical Note 3294). In the first of these two reports, general concepts regarding the effects of the condition of the tire, the type of rolling motion, and the loading are discussed. In Technical Note 3294, the results of a systematic study to determine the effects of temperature and normal pressure on frictional resistance between tire-tread material and concrete are given. Although these data are only a small part of the overall problem, they do offer some insight into the problem of tire-to-runway friction coefficient problem which is being attacked through both experimental flight and laboratory studies at the NACA Langley Laboratory.

In recent years, propeller reversing has been employed very effectively to assist in braking the aircraft during the landing roll on modern propeller-driven aircraft. A similar reverse-thrust device for the modern jet aircraft would be equally useful and can be accomplished by the reversing of the direction of the propulsive jet during landing. The NACA has completed an experimental investigation in which three types of thrust reversers were studied. Models of a target type, a tailpipe cascade type, and a ring cascade type were tested and the effects of design variables on performance and reversed-flow boundaries were determined. This work was reported in Technical Note 3664 and the results indicate that reverse-thrust ratios of from 40 to 80 percent could be obtained and that all three types had satisfactory thrust-modulation characteristics. Performance and operational studies of a full-scale jet engine thrust reverser (Technical Note 3665) of the target type utilized on a turbojet engine were also conducted. This device was pylon mounted under the wing of a cargo airplane to simulate a jet transport. The thrust reverser was operated for both stationary and taxi conditions, but the airplane was not flown. In addition to obtaining the performance of the thrust reverser, heat-rise patterns and rates resulting from impingement of the reversed hot gases on a simulated lower wing were also measured during periods of thrust reversal. Reingestion of the reversed hot gases into the engine inlet constituted an additional operating problem in that the temperature levels were raised throughout the engine and the reversed-thrust ratio was reduced. Taxi tests indicated that at ground speeds of 62 knots, the free-stream velocity was sufficient to prevent the reversed gas flow from entering the engine inlet.

Fire

The NACA has continued its research with turbojet and turboprop types of engines into the problems of crash-fire inerting and, in addition, has been studying the problems of flight fires in jet aircraft. Effective fire-fighting methods are still an important aspect of the problem. At the heart of the problem is the need for potent fire-extinguishing agents which have properties that make them suitable for use on aircraft. In Technical Note 3565, the results of a study which explains the quenching action of halogenated agents in the fire-extinguishing process are given. It is concluded that the presence of halogen in an agent need not reduce its fire-fighting ability provided that there is enough halogen to make the agent noninflammable. The assumption that halogenated agents act merely by chain-breaking reactions with active particles is consistent with the experimental facts available and will help guide the selection of other halogenated agents for further tests of their fire-fighting properties.

Technical Note 3560 presents the results of an investigation conducted at the University of Cincinnati under the sponsorship of the NACA on the spontaneous ignition of lubricants of reduced inflammability. In the initial phase of the investigation, the spontaneous-ignition characteristics of approximately 50 organic compounds were investigated and observations were made on the effects of structure on ignition. In studying compounds of interest as lubricants, it was found that hydrogenated polyisobutylene showed remarkable resistance to spontaneous ignition. Results indicate that those esters possessing high autoignition temperatures have low molecular weights, while those having low molecular weights in the lubricant range show poor resistance to spontaneous ignition.

Gust Alleviation

Whenever rough air is encountered in flight, the recommended practice is to reduce the speed of the airplane to the design speed for maximum gust intensity. When encountering rough air, the pilot does not always have time or advanced warning so that he can reduce the airspeed to the design speed. In these cases, the distance and maneuvering required to reduce speed may have an important bearing on the loads imposed on the airframe. Technical Note 3613 presents the results of an investigation of the problem of reducing the speed of a jet transport in flight. It was found that the required distance was much greater for a jet transport than for a typical piston-engine transport at the same altitude. The distance was also found to increase with altitude up to the altitude for maximum true airspeed. The increased distance for the jet transport was primarily the result of increased kinetic energy and to a lesser extent that of lower drag coefficients. These results are believed to be qualitatively correct for high-speed transport aircraft. The use of aerodynamic brakes, thrust reversal, or a climbing maneuver has been shown to be effective in reducing the distance required to reach the rough-air speed.

The ability of the human pilot to fly a precision course in rough air has been questioned and compared with the ability of an airplane autopilot combination to do the same task. Although the NACA has not studied this question directly, it has conducted theoretical studies involving various types of autopilots in an attempt to learn the characteristics of airplane response to gusts. The results of two such investigations have been published in Technical Notes 3635 and 3603. The results given in the former report indicate that the response to side gusts can be noticeably reduced. In the latter report, when the airplane was flown by various autopilots, the increased yaw damping greatly reduced the resonance associated with Dutch-roll of the airplane. The addition of an autopilot supplied directional stability and roll stability and greatly reduced the yaw and roll responses to gusts. Autopilots that held side forces to low values and provided good course response to command signals allowed large roll response to side gusts.

The NACA has been studying various means of increasing the smoothness of flight through rough air, both theoretically and experimentally. One of the most promising methods utilizes an autocontrol system in which the flaps and elevators are operated in accordance with indications of changes in angle of attack to maintain constant lift and zero pitching moment of the airplane. A detailed analysis of this system and its various refinements is presented in Technical Note 3597, including a study of the transient response of the airplane for both gust disturbances and longitudinal control inputs. The aerodynamic characteristics necessary for optimum gust alleviation are derived and the response of this system is compared with that of the basic airplane. In order to study these analytical results in flight, an experimental investigation was conducted with a light transport airplane whose controls were modified to the extent necessary to provide gust alleviation. The results indicate that the gust alleviation system is at least capable of alleviating the normal acceleration due to gusts by 50 percent at a frequency of 0.6 cycle per second, the natural frequency of the airplane, and by 40 percent at a frequency of 2 cycles per second. The airplane can be controlled adequately when this gust alleviation system is operating.

Other devices that have been proposed for reducing the acceleration effects of rough air are spoilers, deflectors, and spoiler-deflector combinations. These devices have been investigated for both swept- and unswept-wing models. The results have been reported in Technical Note 3705 and it would appear from gust-tunnel and wind-tunnel tests that a forward-located fixed deflector would be a practical and effective alleviator of gust loads on an airplane having unswept wings. Preliminary results on a model having a 35°-swept-back wing have indicated that deflectors, in order to have the same effectiveness as reported for the unswept wing model, would have to be located more to the rearward on the swept wing and would possibly require larger projections if they are to have the same effectiveness they had on the unswept-wing airplane.

Optimum Flight Paths

The climb of turbojet aircraft, and the effects of tangential accelerations, have been analytically determined for minimum time of climb, climb with minimum fuel consumption, and steepest climb. For each flight condition, the optimum Mach number was obtained from the solution of a sixth-order equation whose coefficients are functions of two fundamental parameters: the ratio of minimum drag in level flight to the thrust and the Mach number which represents the flight at constant altitude and maximum lift-drag ratio. Diagrams have been prepared for the quick calculation of the optimum Mach numbers and the effect of acceleration on the rate of climb in tropospheric and stratospheric flight.

Airspeed Measuring Systems

Accurate determination of Mach number is fundamental to any detailed flight research and is of particular importance in the transonic speed range where many of the aerodynamic parameters vary markedly with Mach number. In order to conduct extensive research in this speed range, it was necessary that a suitable airspeed system be devised. Accordingly, calibrations of four airspeed systems installed in a turbojet fighter were determined in flight at Mach numbers up to 1.04 by the NACA radar-phototheodolite method (Technical Note 3526). The results indicate that, of the systems investigated (a nose boom, two different wing-tip booms, and a fuselage-mounted service system), the nose-boom installation is the most suitable for research use at transonic and low supersonic speeds. The static-pressure error of the nose-boom system is small and constant above a Mach number of 1.03 after passage of the fuselage bow shock wave over the airspeed head.

The need for design information to provide rigid tubes to measure total head pressures correctly at high angles of attack and at high speeds has arisen because of the development of airplanes having good maneuverability at supersonic speeds. Conventional tubes, both rigid and swiveling, are unsatisfactory under these conditions. In Technical Note 3641, the results of wind-tunnel tests of 54 total-pressure tubes have been summarized and data are presented on the effects of inclination of the airstream on measured pressures at subsonic, transonic, and supersonic speeds. These data are in a form which permits a more detailed comparison of the effects of pertinent design values.

Spin Hazards and Recovery

The pilot's loss of orientation during spins, especially during unintentional spins, is a rising problem and has apparently led to a number of recent accidents and near-accidents with both trainer and fighter aircraft during acrobatic maneuvers and after recovery from erect spins. In Technical Note 3531, the nature of inverted spins, the optimum control technique for recovery, and some of the apparent reasons for a pilot's loss of orientation are discussed. It is pointed out that a pilot in an inverted spin should attempt to orient himself with respect to direction of turn by referring to the airplane rate-of-turn indicator in order to determine properly the direction of the yawing component of the total spin rotation. Optimum recovery from the inverted spin should then be obtainable by rapidly reversing the rudder from full with this yawing rotation to full against it while the control stick is held full forward and laterally neutral and, shortly thereafter, the stick should be moved from full forward to full back while it is maintained laterally neutral.

The general policy for recovery from either intentional or accidental spins has been to cut off power as soon as possible after the spin is initiated, because of possible adverse effects. In some instances however, pilots have flown out of an otherwise uncontrollable spin by application of full power in a propeller-driven airplane. Such results from power-on spins may have been due to increased effectiveness of the controls in the slipstream. For a jet engine, however, the situation is different, and unpublished data indicate that thrust alone might be of little assistance. For both propeller-driven and jet-propelled airplanes, spin and spin-recovery characteristics may differ for power-on and power-off conditions, as well as for power-on spins to the right and to the left. These differences may at times have caused serious difficulty in recovering from spins in one direction, whereas recoveries from spins in the other direction could be readily achieved. The differences in spins and recoveries may have been due, in part, to the gyroscopic moments produced by rotating propellers or rotating parts of jet engines. For a jet-propelled airplane, the rotating parts of the engine may continue to rotate at nearly full speed for a long time after power is cut off. A preliminary investigation has been made and reported in Technical Note 3480 to determine the gyroscopic effects of jet-engine rotating parts on the erect spin and recovery characteristics of a model of a military attack airplane. Results indicate that rotating parts affected the spin characteristics differently depending on the type of mass load distribution and the direction of spin.

Control Device for Personal-Owner-Type Airplanes

Although most present-day personal-owner-type airplanes possess a slight degree of inherent spiral stability in cruising flight, they show unstable spiral tendencies under operational conditions. The main reasons for this apparent spiral instability are a lack of means for trimming the airplane laterally and directionally. A variation of lateral and directional trim with airspeed and control-system friction prevents the control surfaces from returning to trim position after control deflection. The specific problems facing the pilot of a personal-owner-type airplane are that of maintaining the airplane in wings-level position during times when there is no natural horizon reference and that of keeping the airplane from diverging spirally while he may be preoccupied with navigational problems. It has been demonstrated that the pilot's sense of orientation is unreliable in the absence of a visual reference, as may be the case when inadvertently or unavoidably encountering instrument weather. Technical Note 3637 describes the results of a flight investigation to determine the effectiveness of an automatic aileron trim control device installed in a personal-owner-type airplane. The results indicate that the device is capable of maintaining the airplane in equilibrium over its operational speed range under directional out-of-trim conditions that would cause rapid divergence of the basic airplane. The device also prevents excessive heading wander and airplane gyration in turbulent air without pilot control. A means is provided for holding the airplane in a stabilized turn to facilitate mild maneuvering through the use of the automatic control.

Precision of Instrument Flight in Helicopters

Early studies of helicopter instrument flight indicated the need for improved handling qualities, particularly for low-speed flight and for precision maneuvers such as instrument approaches, sonar dipping, or hoist operations. Although a number of stability parameters affect the handling characteristics, damping about the principal axes appeared to be a worthwhile subject for initial study. Technical Note 3537 presents the results of a study of the effects of increased damping in roll, pitch, and yaw on the instrument flight-handling qualities of a single-rotor helicopter. Electronic components were used to vary the damping of the helicopter, and these variations were evaluated by performing precision maneuvers while flying on instruments. The studies indicated that, for a representative single-rotor helicopter, increased damping can improve the accuracy of the maneuvers and reduce the effort required of the pilot, particularly at low forward speeds. For the speed range considered (25 to 65 knots), increased damping in roll was found to be particularly effective, much more effective than corresponding changes in yaw and pitch.

AERONAUTICAL METEOROLOGY

Atmospheric Turbulence

Previously evaluated effective gust velocities, U_e, from the data available for both convective and frontal types of thunderstorms have been converted to the recently defined derived gust velocities, U_{de}, which take into account the variations with altitude of the airplane response to gusts. The results, given in Technical Note 3538, indicate that the intensities of the derived gust velocities are essentially constant for altitudes up to approximately 20,000 feet in thunderstorms and that an approximate 10-percent reduction in the intensity occurs as altitude is increased to 30,000 feet.

The NACA provided the Cambridge Research Center of the Air Force with a VGH recorder for measuring turbulence in a flight investigation of the jet stream and the Sierra Mountain Wave. Evaluation of the data showed that the turbulence encountered during the flights was generally light.

A review was made of available information concerning continuous operation of airplanes through rough air at low altitudes and high speeds. From the standpoints of crew efficiency and flight precision, it appears that reductions in the loads and motions due to turbulent air to about one-third of those for present operational airplanes may be required for low-altitude flight. A study of design features indicated that the major factors that effect such reductions are increased wing loadings and reduced lift-curve slopes. Reductions in lift-curve slopes accompany low-aspect-ratio, swept-back, and flexible wings. Changes in stability for airplanes with satisfactory stability characteristics were not significant when the loads were changed in rough air. For configurations with low damping, which causes amplification of the loads in continuous turbulence, the use of augmented damping can result in significant reductions in loads.

Characteristics of Icing Cloud

A statistical survey and a preliminary analysis were made of icing data collected from scheduled flights over the United States and Canada from November 1951 to June 1952 by airline aircraft equipped with NACA pressure-type icing-rate meters. Over 600 icing encounters were logged by three airlines operating in the United States, one operating in Canada, and one operating up the Pacific Coast to Alaska. The data provide relative frequencies of icing cloud variables such as horizontal extent of icing, vertical thickness of icing clouds, air temperature, icing rate, liquid-water content, and total ice accumulation.

Liquid-water contents were higher than those from earlier research flights in layer-type clouds but slightly lower than previous ones from cumulus clouds. Broken-cloud conditions, indicated by intermittent icing, accounted for nearly one-half of all the icing encounters. About 90 percent of the encounters did not exceed a distance of 120 miles, and continuous icing did not exceed 50 miles for 90 percent of the unbroken conditions. Icing-cloud thicknesses measured during climbs and descents were less than 4,500 feet for 90 percent of the vertical cloud traverses.

ICING PROBLEMS

Droplet Impingment

Experimental studies have been made in the NACA icing tunnel to determine the effect of a flapped truncated airfoil on surface velocity distribution and droplet-impingement rates and limits. A 6-foot-chord NACA 65_1–212 airfoil was cut successively at the 50- and 30-percent-chord stations to produce the truncated airfoil sections, which were equipped with trailing-edge flaps to alter the flow field. The results indicated that the correct use of such airfoils may permit impingement and icing studies to be conducted with full-scale leading-edge sections in existing small icing tunnels.

The paths of icing cloud droplets into two engine inlets have been calculated (Technical Note 3593) for 0° angle of attack and for a wide range of meteorological and flight conditions. In both types of inlets, the inlet air velocity of one being 0.7 that of the other, a prolate ellipsoid of revolution (10 percent thick) represents either part or all of the forebody at the center of an annular inlet to an engine. The configurations can also represent the fuselage of an airplane with side ram-scoop inlets. Results indicated that the amount of water ingested is not sensitive to small changes in shape of the outer wall, that impingement on the cowl (i. e., amount and distribution) is quite sensitive to the physical shape and surface condition of the wall, and that the use of screens and boundary-layer-removal scoops at the entrance requires careful design because of the shadow zone (zero water concentration) and regions of high concentration. In addition, a general concept showed that lowering the inlet velocity ratio lowers the ingestion efficiency.

The impingement characteristics of several other bodies have been obtained from droplet-trajectory calculations. For an NACA 65A004 airfoil at 0° angle of attack, the amount of water in droplet from impinging on the airfoil, the area of droplet impingement, and the rate of droplet impingement per unit area of the airfoil surface were calculated as given in Technical Note 3586. The results were compared with those previously reported for the same airfoil at angles of attack of 4° and 8°.

For a sphere in an ideal fluid flow, droplet impingement data and equations for determining the collection efficiency, the area, and the distribution of impingement have been presented in terms of dimensionless parameters (Technical Note 3587). The range of flight and atmospheric conditions covered in the calculations was extended considerably beyond the range covered by previously reported calculations for a sphere.

A study has also been made of water-droplet impingement on a rectangular half body in a two-dimensional incompressible flow field (Technical Note 3658). Data on collection efficiency and distribution of water-droplet impingement were obtained by means of a mechanical differential analyzer.

Icing Protection

A better understanding of the performance and penalties of pneumatic de-icers can aid in selecting ice-protection systems for aircraft. Accordingly, an evaluation in icing conditions was made of two types of pneumatic de-icers, one having spanwise inflatable tubes and the other having chordwise inflatable tubes (Technical Note 3564). Measurements were made to determine lift, drag, and pitching-moment changes caused by inflation of the de-icer boots and by ice formations on the boots. In order to help determine the aerodynamic effects of size and location of ridge-type ice formations on an airfoil, spanwise spoilers mounted on the airfoil at various chordwise locations were also studied.

A preliminary experimental study was conducted to determine the feasibility of preventing rain from impinging on aircraft windshields by use of a high-velocity jet-air blast. By this means, raindrops are broken up into a multitude of small droplets by the jet-air blast and deflected around the windshield. The deflection appears feasible for flight speeds up to 150 miles per hour for low-angle (35° or less) windshields. However, visibility through the mist generated by raindrop breakup is a problem requiring solution.

RESEARCH PUBLICATIONS

REPORTS

1210. Analysis of Turbulent Heat Transfer, Mass Transfer, and Friction in Smooth Tubes at High Prandtl and Schmidt Numbers. By Robert G. Deissler.
1211. Experimental Investigation of Free-Convection Heat Transfer in Vertical Tube at Large Grashof Numbers. By E. R. G. Eckert and A. J. Diaguila.
1212. Analog Study of Interacting and Noninteracting Multiple-Loop Control Systems for Turbojet Engines. By George J. Pack and W. E. Phillips, Jr.
1213. Minimum-Drag Ducted and Pointed Bodies of Revolution Based on Linearized Supersonic Theory. By Hermon M. Parker.
1214. Statistical Measurements of Contact Conditions of 478 Transport-Airplane Landings During Routine Daytime Operations. By Norman S. Silsby.
1215. Impingement of Cloud Droplets on a Cylinder and Procedure for Measuring Liquid-Water Content and Droplet Sizes in Supercooled Clouds by Rotating Multicylinder Method. By R. J. Brun, W. Lewis, P. J. Perkins, and J. S. Serafini.
1216. Charts for Estimating Tail-Rotor Contribution to Helicopter Directional Stability and Control in Low-Speed Flight. By Kenneth B. Amer and Alfred Gessow.
1217. Theoretical Prediction of Pressure Distributions on Nonlifting Airfoils at High Subsonic Speeds. By John R. Spreiter and Alberta Alksne.
1218. Effect of Ground Interference on the Aerodynamic and Flow Characteristics of a 42° Sweptback Wing at Reynolds Numbers up to 6.8×10^6. By G. Chester Furlong and Thomas V. Bollech.
1219. Measurement and Analysis of Wing and Tail Buffeting Loads on a Fighter Airplane. By Wilber B. Huston and T. H. Skopinski.
1220. Calculations of Laminar Heat Transfer Around Cylinders of Arbitrary Cross Section and Transpiration-Cooled Walls With Application to Turbine Blade Cooling. By E. R. G. Eckert and J. N. B. Livingood.
1221. Theoretical Study of the Tunnel-Boundary Lift Interference Due to Slotted Walls in the Presence of the Trailing-Vortex System of a Lifting Model. By Clarence W. Matthews.
1222. A Free-Flight Wind Tunnel for Aerodynamic Testing at Hypersonic Speeds. By Alvin Seiff.
1223. Theoretical and Experimental Investigation of Heat Transfer by Laminar Natural Convection Between Parallel Plates. By A. F. Lietzke.
1224. Effects of Wing Position and Fuselage Size on the Low-Speed Static and Rolling Stability Characteristics of a Delta-Wing Model. By Alex Goodman and David F. Thomas, Jr.
1225. Determination of Lateral-Stability Derivatives and Transfer-Function Coefficients from Frequency-Response Data for Lateral Motions. By James J. Donegan, Samuel W. Robinson, Jr., and Ordway B. Gates, Jr.
1226. A Method for the Design of Sweptback Wings Warped to Produce Specified Flight Characteristics at Supersonic Speeds. By Warren A. Tucker.
1227. An Investigation of the Maximum Lift of Wings at Supersonic Speeds. By James J. Gallagher and James N. Mueller.
1228. Calculated Spanwise Lift Distributions, Influence Functions, and Influence Coefficients for Unswept Wings in Subsonic Flow. By Franklin W. Diederich and Martin Zlotnick.
1229. Exact Solutions of Laminar-Boundary-Layer Equations with Constant Property Values for Porous Wall with Variable Temperature. By Patrick L. Donoughe and John N. B. Livingood.
1230. Generalized Indicial Forces on Deforming Rectangular Wings in Supersonic Flight. By Harvard Lomax, Franklyn B. Fuller, and Loma Sluder.
1231. NACA Transonic Wind-Tunnel Test Sections. By Ray H. Wright and Vernon G. Ward.
1232. A Theoretical and Experimental Investigation of the Lift and Drag Characteristics of Hydrofoils at Subcritical and Supercritical Speeds. By Kenneth L. Wadlin, Charles L. Shuford, Jr., and John R. McGehee.
1233. Shock-Turbulence Interaction and the Generation of Noise. By H. S. Ribner.
1234. On the Kernel Function of the Integral Equation Relating the Lift and Downwash Distributions of Oscillating Finite Wings in Subsonic Flow. By Charles E. Watkins, Harry L. Runyan, and Donald S. Woolston.
1235. Standard Atmosphere—Tables and Data for Altitudes to 65,800 feet. By International Civil Aviation Organization and Langley Aeronautical Laboratory.
1236. Arrangement of Fusiform Bodies to Reduce the Wave Drag at Supersonic Speeds. By Morris D. Friedman and Doris Cohen.
1237. A Flight Evaluation of the Longitudinal Stability Characteristics Associated with the Pitch-Up of a Swept-Wing Airplane in Maneuvering Flight at Transonic Speeds. By Seth B. Anderson and Richard S. Bray.
1238. Investigation at Supersonic Speeds of 22 Triangular Wings Representing Two Airfoil Sections for Each of 11 Apex Angles. By Eugene S. Love.
1239. Error in Airspeed Measurement Due to the Static-Pressure Field Ahead of an Airplane at Transonic Speeds. By Thomas C. O'Bryan, Edward C. B. Danforth, and J. Ford Johnston.
1240. An Investigation of the Effects of Heat Transfer on Boundary-Layer Transition on a Parabolic Body of Revolution (NACA RM-10) at a Mach Number of 1.61. By K. R. Czarnecki and Archibald R. Sinclair.
1241. Theoretical and Analog Studies of the Effects of Nonlinear Stability Derivatives on the Longitudinal Motions of an Aircraft in Response to Step Control Deflections and to the Influence of Proportional Automatic Control. By Howard J. Curfman, Jr.
1242. Transonic Flow Past Cone Cylinders. By George E. Solomon.
1243. High-Resolution Autoradiography. By George C. Towe, Henry J. Gomberg, and J. W. Freeman.
1244. Free-Stream Boundaries of Turbulent Flows. By Stanley Corrsin and Alan L. Kistler.
1245. Analysis and Calculation by Integral Methods of Laminar Compressible Boundary Layer With Heat Transfer and

With and Without Pressure Gradient. By Morris Morduchow.

1246. The Hydrodynamic Characteristics of Modified Rectangular Flat Plates Having Aspect Ratios of 1.00, 0.25, and 0.125 and Operating Near a Free Water Surface. By Kenneth L. Wadlin, John A. Ramsen, and Victor L. Vaughan, Jr.

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1248. An Experimental Study of Applied Ground Loads in Landing. By Benjamin Milwitzky, Dean C. Lindquist, and Dexter M. Potter.

1249. A Unified Two-Dimensional Approach to the Calculation of Three-Dimensional Hypersonic Flows, With Application to Bodies of Revolution. By A. J. Eggers, Jr., and Raymond C. Savin.

1250. The Dynamic-Response Characteristics of a 35° Swept-Wing Airplane as Determined From Flight Measurements. By William C. Triplett, Stuart C. Brown, and G. Allen Smith.

1251. Stress Analysis of Circular Semimonocoque Cylinders With Cutouts. By Harvey G. McComb, Jr.

1252. Quasi-Cylindrical Theory of Wing-Body Interference at Supersonic Speeds and Comparison With Experiment. By Jack N. Nielsen.

1253. A Correlation by Means of Transonic Similarity Rules of Experimentally Determined Characteristics of a Series of Symmetrical and Cambered Wings of Rectangular Plan Form. By John B. McDevitt.

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3218. Flight Determination of the Drag and Pressure Recovery of an NACA 1–40–250 Nose Inlet at Mach Numbers From 0.9 to 1.8. By R. I. Sears and C. F. Merlet.

3271. Thermodynamic Properties of Gaseous Nitrogen. By Harold W. Woolley.

3272. Generalized Tables of Corrections to Thermodynamic Properties for Nonpolar Gases. By Harold W. Woolley and William S. Benedict.

3274. Some Linear Dynamics of Two-Spool Turbo Jet Engines. By David Novik.

3275. Investigation of the Effect of Impact Damage on Fatigue Strength of Jet-Engine Compressor Rotor Blades. By Albert Kaufman and André J. Meyer, Jr.

3277. Space Heating Rates for Some Premixed Turbulent Propane-Air Flames. By Burton D. Fine and Paul Wagner.

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Part II—COMMITTEE ORGANIZATION AND MEMBERSHIP

The National Advisory Committee for Aeronautics was established by Act of Congress approved March 3, 1915 (U. S. Code, title 50, sec. 151). The Committee consists of seventeen members appointed by the President, and includes two representatives each of the Department of the Air Force, the Department of the Navy, and the Civil Aeronautics Authority; one representative each of the Smithsonian Institution, the United States Weather Bureau, and the National Bureau of Standards; and "one Department of Defense representative who is acquainted with the needs of aeronautical research and development." In addition seven members are appointed for five-year terms from persons "acquainted with the needs of aeronautical science, either civil or military, or skilled in aeronautical engineering or its allied sciences." The representatives of the Government organizations serve for indefinite periods, and all members serve as such without compensation.

The following changes in membership have taken place during the past year:

The Committee lost a valuable member by the death on January 4, 1956, of Mr. Ralph S. Damon, President of Trans World Air Lines, Inc., who had been serving as Chairman of the important NACA Committee on Operating Problems. In its tribute to Mr. Damon's memory, the NACA at its meeting on January 19, 1956, said: "His intelligence, enthusiasm, sound judgment, and high qualities of integrity and sincerity, together with his wealth of experience, enabled him to support most effectively the responsibilities of the Committee and to provide highly competent leadership."

To succeed Mr. Damon, President Eisenhower on April 14, 1956, appointed the well-known World War I ace and aviation executive, Captain Edward V. Rickenbacker, Chairman of the Board of Eastern Air Lines, Inc., to membership on the NACA.

On January 6, 1956, the President appointed Hon. Clifford C. Furnas, Assistant Secretary of Defense (Research and Development), a member of NACA. Dr. Furnas succeeded Hon. Donald A. Quarles, Secretary of the Air Force, who had previously served in Dr. Furnas' present post in the Department of Defense.

Vice Admiral William V. Davis, USN, Deputy Chief of Naval Operations (Air), was appointed a member of the NACA on August 2, 1956, succeeding Vice Admiral Thomas S. Combs, who had just been detached from the same Navy post and assigned to other duty.

In accordance with the regulations of the Committee as approved by the President, the chairman and vice chairman and the chairman and vice chairman of the Executive Committee are elected annually.

Prior to the annual meeting of the NACA on October 17, 1956, Dr. Jerome C. Hunsaker, who had been chairman since August 1941, indicated his desire to retire from the chairmanship of the NACA and of the Executive Committee. At the meeting the NACA elected Dr. James H. Doolittle chairman of the NACA and of the Executive Committee. Dr. Leonard Carmichael was re-elected vice chairman of the NACA and Dr. Detlev W. Bronk vice chairman of the Executive Committee.

The Committee membership is as follows:

James H. Doolittle, Sc. D., Shell Oil Company, Chairman.
Leonard Carmichael, Ph. D., Secretary, Smithsonian Institution, Vice Chairman.
Joseph P. Adams, LL. B., Vice Chairman, Civil Aeronautics Board.
Allen V. Astin, Ph. D., Director, National Bureau of Standards.
Preston R. Bassett, M. A., Vice President, Sperry Rand Corporation.
Detlev W. Bronk, Ph. D., President, Rockefeller Institute for Medical Research.
Frederick C. Crawford, Sc. D., Chairman of the Board, Thompson Products, Inc.
William V. Davis, Jr., Vice Admiral, United States Navy, Deputy Chief of Naval Operations (Air).
Clifford C. Furnas, Ph. D., Assistant Secretary of Defense (Research and Development).
Jerome C. Hunsaker, Sc. D., Massachusetts Institute of Technology.
Carl J. Pfingstag, Rear Admiral, United States Navy, Assistant Chief for Field Activities, Bureau of Aeronautics.
Donald L. Putt, Lieutenant General, United States Air Force, Deputy Chief of Staff, Development.
Arthur E. Raymond, Sc. D., Vice President—Engineering, Douglas Aircraft Company, Inc.
Francis W. Reichelderfer, Sc. D., Chief, United States Weather Bureau.
Edward V. Rickenbacker, Sc. D., Chairman of the Board, Eastern Air Lines, Inc.
Louis S. Rothschild, Under Secretary of Commerce for Transportation.
Nathan F. Twining, General, United States Air Force, Chief of Staff.

Assisting the Committee in its coordination of aeronautical research and the formulation of its research programs are four main technical committees: Aerodynamics, Power Plants for Aircraft, Aircraft Construction, and Operating Problems. Each of these committees is assisted by four or more subcommittees. Effective January 1, 1956, two new subcommittees were established under the Committee on Aerodynamics in place of the Subcommittee on Stability and Control, namely: Aerodynamic Stability and Control, and Automatic Stabilization and Control. This action was taken because of the increase in the importance of the problems of automatic stabilization and control in connection with both piloted aircraft and missiles.

The Committee is advised on matters of policy affecting the aircraft industry by an Industry Consulting Committee.

The membership of the committees and their subcommittees is as follows:

COMMITTEE ON AERODYNAMICS

Mr. Preston R. Bassett, Vice President, The Sperry Rand Corp., Chairman.
Dr. Theodore P. Wright, Vice President for Research, Cornell University, Vice Chairman.
Col. Daniel D. McKee, USAF, Wright Air Development Center.
Rear Adm. W. A. Schoech, USN, Assistant Chief of the Bureau of Aeronautics for Research and Development, Department of the Navy.
Mr. F. A. Louden, Bureau of Aeronautics, Department of the Navy.
Dr. H. H. Kurzweg, Associate Technical Director for Aeroballistic Research, Naval Ordnance Laboratory.
Maj. Gen August Schomburg, USA, Assistant Chief of Ordnance for Research and Development, Department of the Army.
Mr. D. M. Thompson, Office of the Chief of Transportation, Department of the Army.
Mr. Harold D. Hoekstra, Civil Aeronautics Administration.
Dr. Hugh L. Dryden (ex officio).
Mr. Floyd L. Thompson, NACA Langley Aeronautical Laboratory.
Mr. Russell G. Robinson, NACA Ames Aeronautical Laboratory.
Capt. W. S. Diehl, USN (Ret.).
Mr. L. L. Douglas, Vice President—Engineering, VERTOL Aircraft Corp.
Rear Adm. R. S. Hatcher, USN (Ret.), Professor and Chairman, Department of Aeronautical Engineering, New York University.
Mr. Clarence L. Johnson, Chief Engineer, Lockheed Aircraft Corp.
Dr. A. Kartveli, Vice President—Engineering, Republic Aviation Corp.
Mr. Schuyler Kleinhans, Assistant Chief Engineer, Santa Monica Division, Douglas Aircraft Co.
Dr. Albert E. Lombard, Jr., Director of Research, McDonnell Aircraft Corp.
Dr. Clark B. Millikan, Director, Daniel Guggenheim Aeronautical Laboratory, California Institute of Technology.
Dr. William J. O'Donnell, Assistant Chief Engineer—Development and Experimental, Republic Aviation Corp.
Mr. Kendall Perkins, Vice President—Engineering, McDonnell Aircraft Corp.
Mr. H. A. Storms, Jr., Chief, Technical Engineering, North American Aviation, Inc.
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Mr. George S. Trimble, Jr., Vice President—Chief Engineer, The Martin Co.
Mr. Robert J. Woods, Chief Design Engineer, Bell Aircraft Corp.

Mr. Milton B. Ames, Jr., Secretary

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Mr. E. Haynes, Air Research and Development Command.
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Mr. John D. Nicolaides, Bureau of Ordnance, Department of the Navy.
Dr. Joseph Sternberg, Ballistic Research Laboratories, Aberdeen Proving Ground.
Dr. G. B. Schubauer, Chief, Fluid Mechanics Section, National Bureau of Standards.
Dr. Adolf Busemann, NACA Langley Aeronautical Laboratory.
Mr. John Stack, NACA Langley Aeronautical Laboratory.
Dr. D. R. Chapman, NACA Ames Aeronautical Laboratory.
Mr. Robert T. Jones, NACA Ames Aeronautical Laboratory.
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Prof. Walker Bleakney, Princeton University.
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Mr. A. S. Irwin, Assistant Chief Engineer—Aircraft, Marlin-Rockwell Corp.
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Dr. George F. Wislicenus, Director, Garfield Thomas Water Tunnel, Ordnance Research Laboratory, The Pennsylvania State University.

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Prof. Pol Duwez, California Institute of Technology.
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Mr. L. R. Jackson, Assistant Director, Battelle Memorial Institute.
Dr. J. C. McDonald, Assistant Technical Director, Magnesium Department, The Dow Chemical Co.
Mr. Peter Payson, Assistant Director of Research, Crucible Steel Company of America.
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Dr. Ceorge V. Smith, Professor of Metallurgical Engineering, Cornell University.

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Mr. Arthur E. Smith, Engineering Manager, Pratt & Whitney Aircraft, United Aircraft Corp.
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Mr. H. W. Withington, Staff Engineer—Systems, Boeing Airplane Co.

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Brig. Gen. Thomas S. Moorman, Jr., USAF, Commander, Air Weather Service.
Mr. C. N. Touart, Chief, Atmospheric Analysis Laboratory, Geophysics Research Directorate, Air Force Cambridge Research Center.
Capt. Paul R. Drouilhet, USN, Head, Naval Aerology Branch, Office of Chief of Naval Operations.
Mrs. Frances L. Whedon, Office of the Chief Signal Officer, Department of the Army.
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Dr. Harry Wexler, Chief, Scientific Services Division, U. S. Weather Bureau.
Mr. Robert W. Craig, Civil Aeronautics Administration.
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Mr. G. Donald Linklater, Senior Meteorologist, Northwest Airlines, Inc.
Mr. Bernard L. Messinger, Department Manager, Thermodynamics, Lockheed Aircraft Corp.
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Capt. C. P. Phoebus, (MC) USN, Office of Naval Research, Department of the Navy.
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Mr. Arthur A. Regier, NACA Langley Aeronautical Laboratory.
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Dr. Richard H. Bolt, Massachusetts Institute of Technology.
Mr. Allen W. Dallas, Director, Engineering Division, Air Transport Association of America.
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Part III—FINANCIAL REPORT

Funds appropriated for the Committee for the fiscal years 1956 and 1957 and obligations against the fiscal year 1956 appropriations are as follows:

	Fiscal year 1956		Fiscal year 1957
	Allotments	Obligations	Allotments
SALARIES AND EXPENSES APPROPRIATION			
NACA Headquarters	$1, 557, 745	$1, 541, 237	$1, 624, 050
Langley Aeronautical Laboratory	22, 141, 400	22, 051, 384	23, 778, 100
Ames Aeronautical Laboratory	10, 929, 250	10, 850, 663	12, 978, 600
Lewis Flight Propulsion Laboratory	20, 237, 805	20, 200, 066	21, 591, 244
High-Speed Flight Station	1, 929, 695	1, 913, 134	2, 090, 950
Pilotless Aircraft Station	928, 500	908, 622	1, 095, 335
Western Coordination Office	23, 935	19, 979	32, 105
Wright-Patterson Liaison Office	15, 604	15, 439	16, 116
Research contracts with educational institutions	750, 300	750, 291	770, 000
Research contracts with Government agencies	198, 000	198, 000	200, 000
Savings reserved for reappropriation	1, 422, 766	1, 500, 000	
Unobligated balance		186, 185	
Total	[1] 60, 135, 000	60, 135, 000	[2] 64, 176, 500
CONSTRUCTION AND EQUIPMENT APPROPRIATION			
Langley Aeronautical Laboratory	3, 325, 000	31, 741	7, 826, 000
Ames Aeronautical Laboratory	1, 055, 000	418, 898	906, 000
Lewis Flight Propulsion Laboratory	8, 395, 000	1, 796, 349	5, 712, 000
Pilotless Aircraft Station	90, 000	1, 595	
Reserve transferred from prior years	−300, 000	−32, 013	−444, 000
Unobligated balance		[3] 10, 348, 430	
Total	[1] 12, 565, 000	12, 565, 000	[2] 14. 000, 000

[1] Appropriated in the Independent Offices Appropriation Act, 1956, approved June 30, 1955.

[2] Appropriated in the Independent Offices Appropriation Act, 1957, approved June 27, 1956, and the First Supplemental Appropriation Act, 1957, approved July 27, 1956. Includes $1,500,000 reappropriation of fiscal year 1956 funds.

[3] This balance remains available until expended.